turned off or has malfunctioned. The BMAGs can also be used to produce attitude rate-of-change information, as backup for the rate gyros.

The BMAGs are single-degree-of-freedom, miniature integrating gyros contained in electrically heated individual packages. The heater will maintain individual BMAG temperature at 170±2°F. Degraded gyro operation will result if this temperature is not maintained. The gyro spin motors are 3-phase 24,000-rpm synchronous devices, powered by 13.6 volts 400 cps from a supply in the attitude gyro accelerometer package electronic control assembly. With C/M temperature at 80°F and the mounting plate temperature at 55°F, the maximum time allowed for the BMAG to reach operating limits is 40 minutes. A few gyro characteristics are listed as follows.

Acceleration sensitive drift	4 deg/hr
Maximum self-torquing rate	25 deg/sec
Maximum attitude range	±20 deg
AGCU coupled torquing limits	20° sec roll, 5° sec pitch and yaw

2.3.4.2.2 Accelerometer.

The accelerometer is mounted along the spacecraft X-axis to sense velocity changes along this axis. It is a pendulous-type accelerometer with electronic null and balance. The temperature is maintained at 170±2°F under normal operating conditions. A signal output is generated by a velocity change along the X-axis. This velocity change causes the pendulous mass to move, resulting in a change of coupling between the primary and secondary windings of a signal generator. This results in an output signal which is demodulated and amplified to provide acceleration information in the form of digital signals to a counter in the delta V display (paragraph 2.3.4.10). The pendulous mass is returned to null by the balancing action of the electronic caging signal when the velocity change ceases.

2.3.4.3 Pitch, Roll, and Yaw Electronic Control Assemblies.

The pitch, yaw, and roll electronic control assemblies (ECAs) are nearly identical, with slight differences due to different requirements for each axis. The ECAs provide the circuitry for input control signal processing and SCS mode control and configuration. Input control signals consist of attitude error signals from the SCS BMAGs and from the G&N system, minimum impulse commands, rate gyro angular signals, translation and rotation control commands, service propulsion engine gimbal position commands, and SPS engine gimbal rate and position feedback signals. Mode control inputs are received from switches on panel 8 of the main display console (MDC). These input signals are applied to logic-controlled relays which enable circuit configurations corresponding to the desired mode. ECA output signals consist of reaction jet firing commands and SPS gimbal position commands. The reaction jet commands are generated in the jet selection logic portions of the ECAs. Preignition SPS engine gimbal position commands are generated manually at the AS/GPI by thumbwheels which provide input signals to the gimbal control circuits. Post-ignition gimbal control is provided by automatic thrust vector control (TVC) circuitry in the pitch and yaw ECAs.

2.3.4.4 Auxiliary Electronic Control Assembly.

The auxiliary electronic control assembly contains the attitude gyro coupling unit and service propulsion system thrust on-off command circuitry. The AGCU portion of the auxiliary ECA receives BMAG signals from the DISPLAY-AGAA ECA and processes the data for display on the FDAI. It also transforms attitude set dial signals to body axes. The SPS engine on-off control circuitry in the AUX ECA receives engine on-off commands from the G&N system or the SCS. These commands are conditioned and applied to the SPS engine solenoid valves.

2.3.4.5 Display and Attitude Gyro Accelerometer Assembly Electronic Control Assembly.

The display and attitude gyro accelerometer assembly electronic control assembly provides the electronic circuitry required to control and power the displays, BMAGs, and accelerometer. The DISPLAY ECA portion consists of the circuitry necessary to receive and condition the following:

- Attitude error signals from the G&N system or the BMAGs to the FDAI attitude error indicators
- Attitude rate-of-change signals from the rate gyros or BMAGs to the FDAI attitude rate indicators
- Feedback signals from the SPS engine gimbals to the gimbal position indicators
- Accelerometer signals from the AGAA to the delta V display integrator.

The AGAA ECA portion consists of circuitry necessary to accomplish the following:

- Accept and condition BMAG inputs for the AGCU
- Accept and condition AGCU torquing commands to the BMAGs
- Control BMAG and accelerometer temperature controls
- Control and condition the accelerometer rebalance loop and inputs to the integrator
- Supply reference voltages to the BMAGs and accelerometer
- Condition the BMAG and accelerometer outputs to telemetry.

2.3.4.6 Rotation Controls.

Two identical rotation controls are provided to enable manual command of the spacecraft attitude (figure 2.3-8). Each control is a control stick containing breakout switches and transducers (figure 2.3-9) which apply control signals to the reaction jet selection logic circuitry in the pitch, yaw, and roll ECAs. When the stick is moved, the breakout switches close the loop between the BMAGs and AGCU, and control signals from the pitch, yaw, and roll transducers are applied to the appropriate control ECAs. Proportional response to the control stick movement is provided by rate gyro feedback to the ECA electronics (figure 2.3-10). The reaction jet solenoids can also be controlled via the direct application of control voltages from switches in the controller. The latter method, requiring use of the DIRECT MODE switch on MDC-8, does not provide proportional response. With the direct mode enabled and the stick commanding a rotation about one axis, the rate damping circuits in the other axes are active.

Provision is made to mount the controls at four different locations in the C/M: the right armrest of the left crew couch, both armrests of the right crew couch, and at the navigation station in the lower equipment bay. Normally, one control is attached to the left crew couch mount and the other is used at any of the other three locations. Simultaneous operation of both controls is possible but not advised; however, each control has a locking device to prevent inadvertent operation.

2.3.4.7 Translation Controls.

Two identical translation controls are provided to enable manual command of spacecraft translational maneuvers. (See figure 2.3-8.) Each control is a T-handle type control stick containing switches, which apply control signals to the reaction jet selection logic circuitry in the pitch, yaw, and roll ECSs (figure 2.3-11). The reaction jets are activated in groups of two or four, depending upon the direction of desired translation. Left-right and up-down translations are accomplished by firing two reaction control jets with the same direction of thrust. Forward and reverse translations are accomplished by firing the four forward thrusting or four rearward thrusting reaction control jets. The translation maneuver commanded is in direct response to the direction in which the T-handle is moved.

The primary control has switches which initiate a spacecraft abort if the T-handle is rotated counterclockwise. Both controls have switches which enable manual thrust vector control and disable automatic attitude control if the T-handle is rotated clockwise. Each control has a locking device to prevent inadvertent operation of the translation control.

Both controls are mounted in the C/M on the left armrest of the crew couch. The primary control is identified with yellow stripes and is the only one provided with abort switching capability.

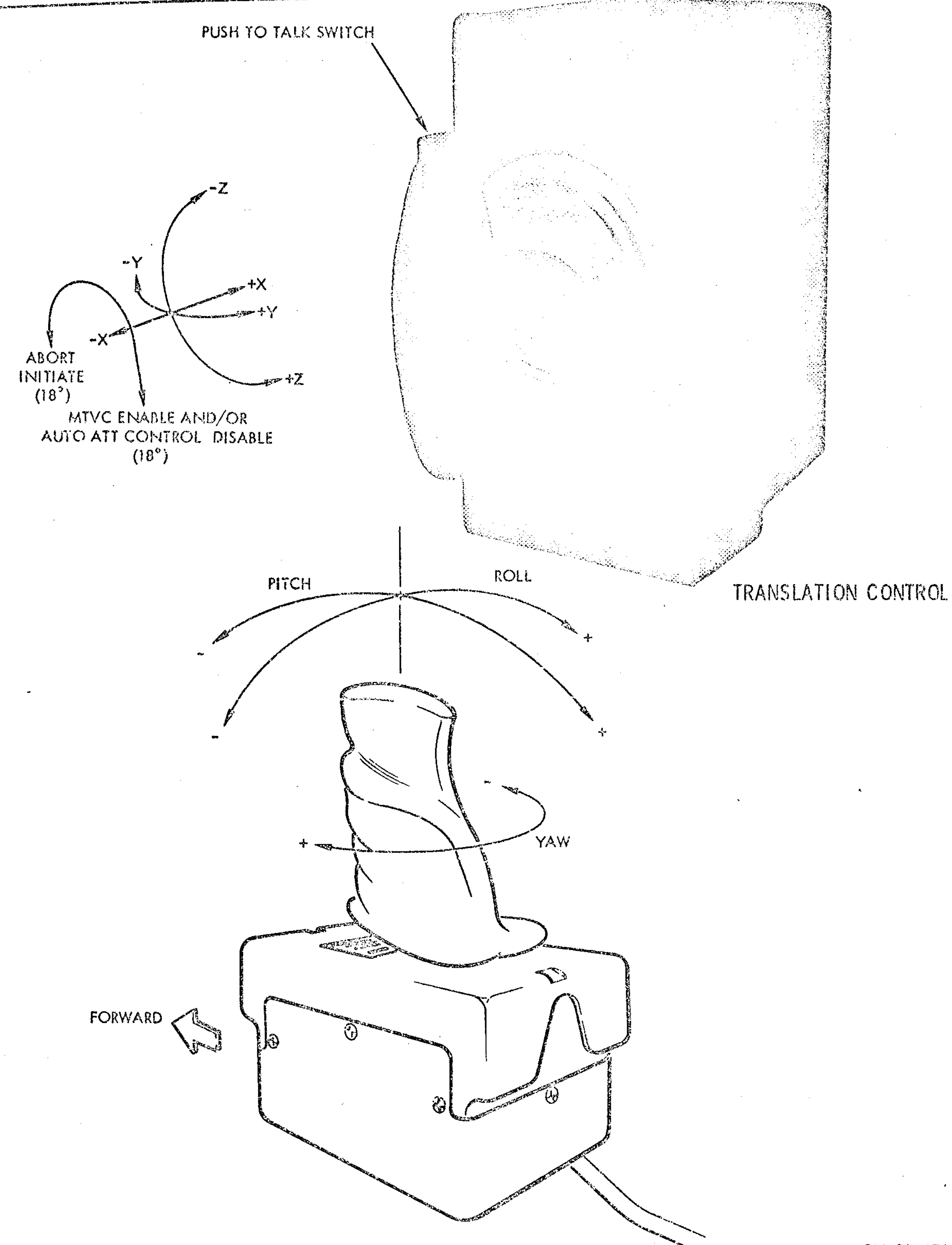
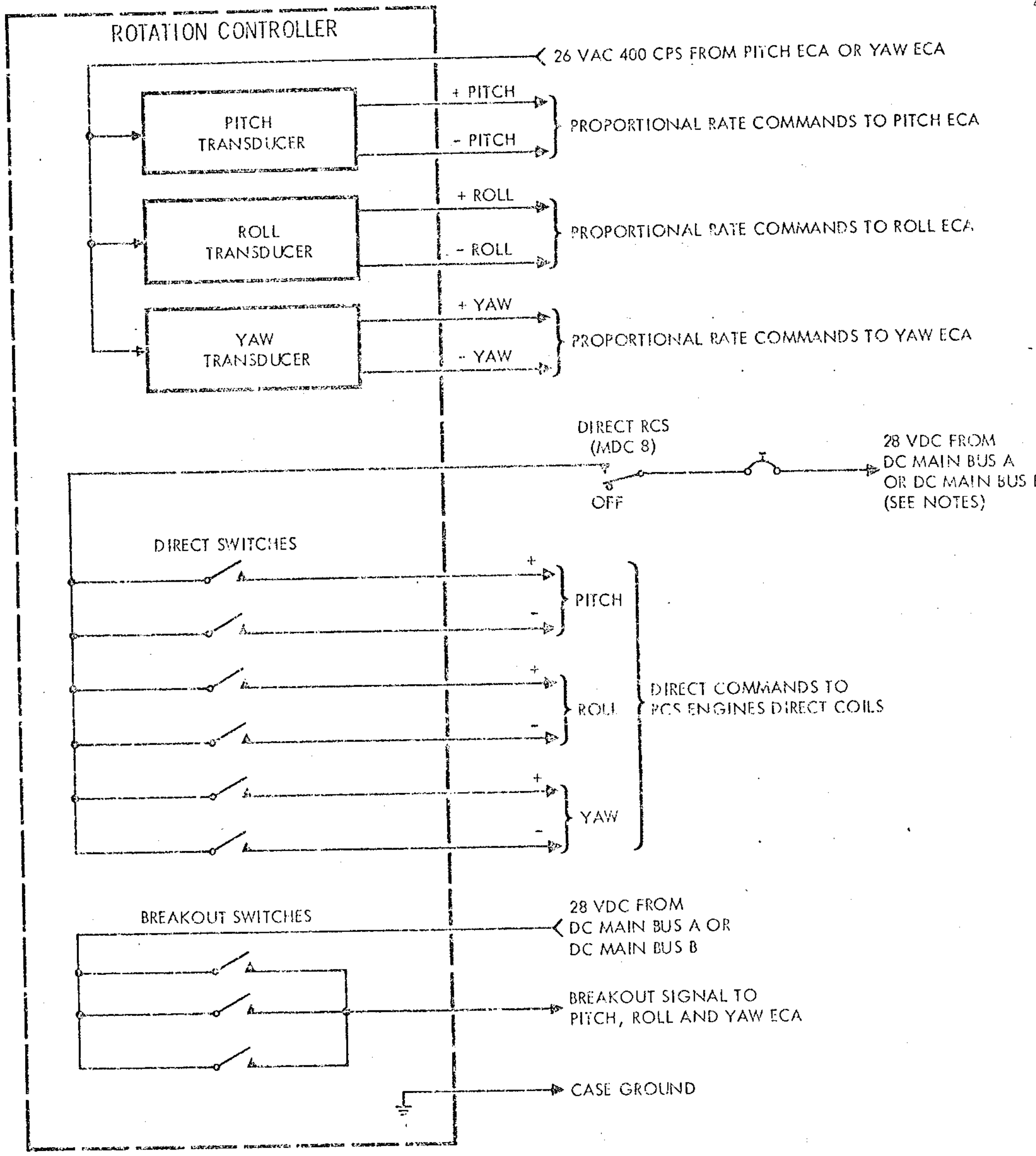


Figure 2.3-8. Rotation and Translation Controls



NOTES:
1. DC main bus A supplies power to direct switches in rotation control No. 1.
2. DC main bus B supplies power to direct switches in rotation control No. 2.

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Figure 2.3-9. Rotation Control Schematic

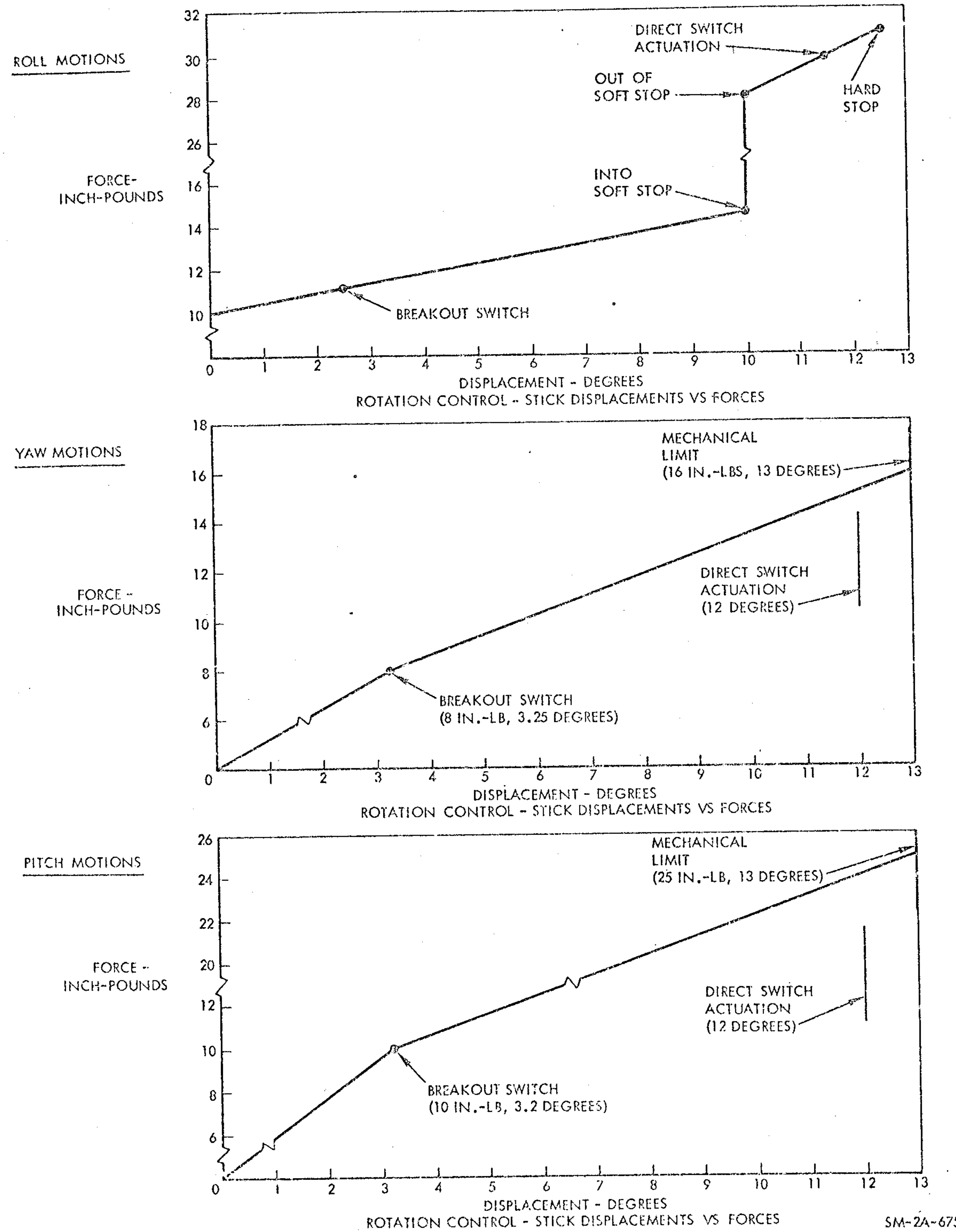


Figure 2.3-10. Rotation Control Charts

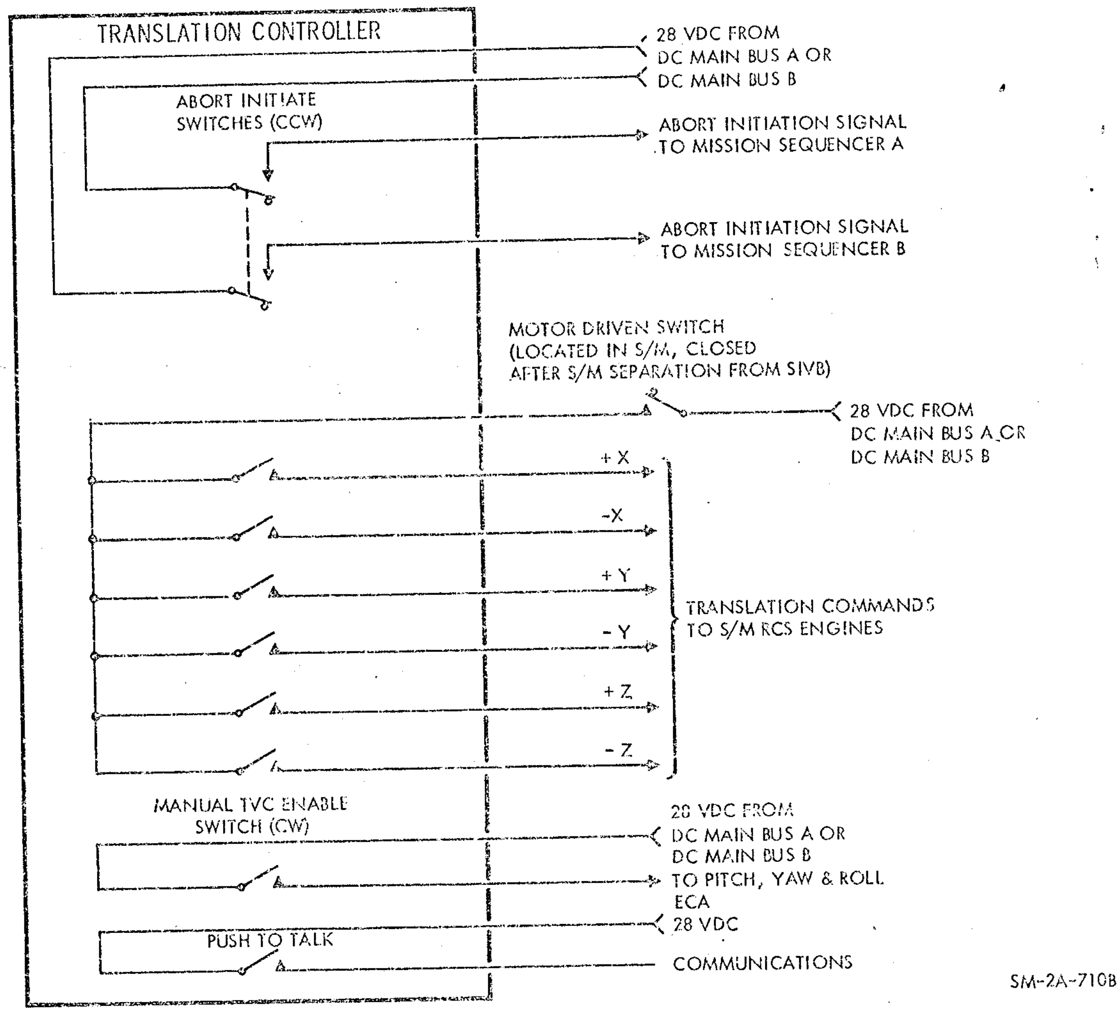
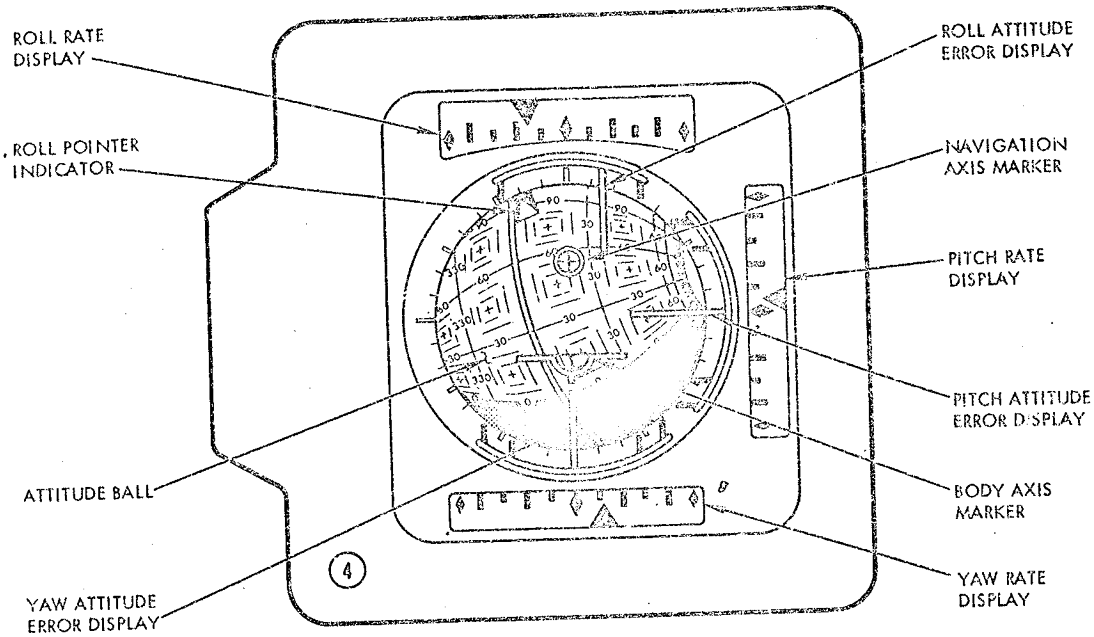


Figure 2.3-11. Translation Control Schematic

2.3.4.8 Flight Director Attitude Indicator.

The flight director attitude indicator, located on MDC-4, provides attitude, attitude error, and attitude rate-of-change display information. (See figure 2.3-12.) The FDAI indicates attitude rate of change and attitude error on indicators and spacecraft attitude on a 3 degree-of-freedom, inertially referenced ball and roll indicator. The roll rate display is located across the top of the FDAI. The pitch rate display is located along the right side of the FDAI. The yaw rate display is located across the bottom of the FDAI. The attitude error displays are pointers which move across the face of the ball. The rate and error displays are fly-to displays. The roll attitude pointer is located at the top and points down towards the center of the ball. The pitch attitude pointer is located at the right and points left toward the center of the ball. The yaw pointer is located at the



DISPLAY	AXIS		
	Pitch (θ)	Yaw (ψ)	Roll (ϕ)
Attitude Ball	+55°	+21°	
Roll "Bug"			+21°
Attitude Error Indicators	0°	0°	-2.5°
Angular Rate Indicators	-.1°/Sec	-.2°/Sec	+.3°/Sec

NOTE: On this illustration, the attitude ball display is read with reference to the navigation axis marker.

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Figure 2.3-12. Flight Director Attitude Indicator

bottom and points up toward the center of the ball. The FDAI is located in the center of the display. The ball face is marked in degrees of pitch and yaw and is referenced to the IMU axes. Two reference markers are located on the face of the display to indicate attitude in reference to either spacecraft body axes or IMU (navigation) axes The body axes marker is on the lower face of the display. The IMU or navigation axes marker is located on the upper face of the display. Roll attitude is the IMU reference and is indicated by the roll indicator (bug) which travels around the circumference of the ball. Roll attitude zero reference is located at the top of the ball.

FDAI display operation will vary, depending upon mode. The rate display is operative at all times. Each rate indicator has a calibrated display scale, with the range of the scale dependent upon the mode selected. Maximum full-scale deflection will be as indicated in the following tabulation.

Mode	Full-Scale Deflection		
	Pitch	Roll	Yaw
Delta V modes	±5°/sec	±5°/sec	±5°/sec
Entry modes and monitor mode	±5°/sec	±25°/sec	±5°/sec
All other modes	±1°/sec	±1°/sec	±1°/sec

The attitude error display function depends upon mode. During attitude hold modes, the attitude error pointers display attitude changes within the selected deadband. The maximum deadband will allow approximately ±5 degrees of spacecraft movement in any axis, which is equivalent to full-scale deflection for the attitude error pointers. This results in easily discernible pointer movement. The minimum deadband allows approximately ±0.5 degrees of spacecraft movement in any axis, which results in very little pointer movement. In SCS control modes, the attitude error signals are derived from the BMAGs or AGCU. The AGCU alone provides the driving signals only when used in conjunction with the ATT SET switch for a dialed attitude change. In G&N modes, the attitude error signals are derived from the G&N coupling display units (CDUs). Full-scale deflection varies depending upon the mode selected, as indicated by the following tabulation.

Mode	Full-Scale Deflection		
	Pitch	Roll	Yaw
Entry modes	±5°	±25°	±5°
Monitor mode	±15°	±25°	±15°
All other modes	±5°	±5°	±5°

The gimbaled attitude ball is colored half gray and half black, with the line of separation at the 0-degree to 180-degree pitch angle. This two-color scheme permits rapid recognition of the specific pitch hemisphere being displayed. There are two concentric circles located at 90-degree yaw/0-degree pitch and 270-degree yaw/0-degree pitch. The inner circles are solid red 30-degree circles which represent the base of a cone with the apex at the center of the ball. This area denotes possible IMU gimbal lock. The outer circles enclose a red 40-degree circle which also represents the base of a cone with the apex at the center of the ball. This area denotes attitudes which result in AGCU inaccuracy. The ball is read against one of two reference marks, depending upon the mode of operation. The IMU axes reference mark is the upper marker (⊕) which indicates spacecraft attitude in pitch and yaw axes with reference to the IMU gimbals or stability axes. The body axes marker is the lower marker (⌐◡¬) which indicates spacecraft attitude with reference to the body axes. The included angle between the markers is 32 degrees. The roll indicator is referenced to the zero mark on the periphery of the ball display face in all operational modes. In SCS modes, the ball is driven only during manual maneuvers, during FDAI alignment, and after 0.05 g is sensed during entry. During these functions, the ball is driven by signals received from the AGCU. In G&N modes, the ball displays IMU gimbal angles. Figure 2.3-13 provides a tabular listing of the various FDAI display configurations.

SCS

FLIGHT CONTROL MODES	SCS Entry	SCS ΔV	SCS Local Vertical	SCS Attitude Control
Total Attitude (Ball)	Before 0.05 Gs, ball driven only during manual maneuver or FDAI alignment; after 0.05 Gs driven continuously.	Ball driven only during manual maneuver or FDAI alignment.	Ball driven only during manual maneuver or FDAI alignment.	Ball driven only during manual maneuver or FDAI alignment.
Attitude Errors	BMAGs P & Y±5° max deflection; R±25° max deflection.	BMAGs P, Y, & R±5° max deflection.	BMAGs P, Y, & R±5° max deflection.	BMAGs P, Y, & R±5° max deflection.
Attitude Rate Errors	Rate gyros (BMAGs when backup rate switch actuated). P & ° 5°/sec deflection. R 25°/sec deflection.	Rate gyros (BMAGs when backup rate switch actuated). P, Y, & R±5°/sec deflection.	Rate gyros (BMAGs when backup switch actuated). P, Y, & R±1°/sec deflection.	Rate gyros (BMAGs when backup rate switch activated). P, Y, & R±1°/sec deflection.
Roll Bug	AGCU roll angle	AGCU roll angle	AGCU roll angle	AGCU roll angle

G&N

FLIGHT CONTROL MODES	G&N Entry	G&N ΔV	G&N Attitude Control	Monitor
Total Attitude (Ball)	Ball repeats IMU position.	Ball repeats IMU position.	Ball repeats IMU position.	Ball repeats IMU position
Attitude Errors	CDU P, Y, ±5° max deflection. R±25° max deflection.	CDU P, Y, & R±5° max deflection.	CDU P, Y, & R ±5° max deflection.	CDU P, Y±15°, & R±25° max deflection.
Attitude Rate Errors	Rate gyros (BMAGs when backup rate switch actuated). P & Y±5°/sec deflection. R±25°/sec deflection.	Rate gyros (BMAGs when backup rate switch actuated). P, Y, & R 5°/sec deflection.	Rate gyros (BMAGs when backup rate switch actuated). P, Y, & R 1°/sec deflection.	Rate gyros (BMAGs when backup rate switch actuated). P, Y, & R 5°/sec deflection.
Roll Bug	IMU roll angle	IMU roll angle	IMU roll angle	IMU roll angle

Figure 2.3-13. FDAI Display Configuration

2.3.4.9 Attitude Set/Gimbal Position Indicator.

The attitude set/gimbal position indicator (AS-GPI) provides two separate functions. The attitude set portion enables FDAI-AGCU alignment and provides a reference for acquiring new attitudes. The gimbal position display portion provides positioning control of the SPS engine gimbal angles and displays gimbal position. The attitude set portion provides a comparison of the angle between the attitude displayed on the ATTITUDE SET indicators and the position of the AGCU resolver shaft. It also provides control of the inputs to the FDAI attitude error pointers and aligns the FDAI to the attitude indicated on the ATTITUDE SET indicators. The attitude set function is controlled by manual operation of the thumbwheels. The desired position is dialed by a thumbwheel, with the display providing visual indication of the position selected by the thumbwheel. The attitude display may be used to align the FDAI-AGCU or to indicate a new attitude for the spacecraft. To accomplish FDAI-AGCU alignment, the desired position is set on the attitude set display. The FDAI ALIGN pushbutton is then pressed. This provides a signal which drives the AGCU stepper motor and control logic to position the AGCU resolver shaft. The stepper motor drives the shaft to the commanded position, resulting in an output from an angle generator to the FDAI ball, driving the ball to the commanded position. To manually maneuver the spacecraft to a predetermined attitude, the display is set to the desired attitude and the ATT SET switch is placed to ATT SET. The output signal is applied to the attitude error display pointers on the FDAI, providing a manual maneuver indication to the space crew. The display configuration is a fly-to type, with the crew manually maneuvering the spacecraft toward the pointers. (The pointers will peg if the new attitude is in excess of 5 degrees from the original attitude.)

The SPS engine gimbal position display allows positioning of the gimbals and provides indication of gimbal position. Two thumbwheels enable the pitch and yaw engine gimbals to be positioned prior to SPS engine firing. The gimbals position the engine to an attitude which ensures that the SPS engine thrust vector is through the spacecraft center of gravity. Movement of the thumbwheels results in the generation of control signals which engage the SPS gimbal ring torque motor magnetic clutches. Clutch engagement connects the gimbal torque motor and gear train to move the gimbal ring. Position transducers send a signal to the gimbal position displays to indicate the angle of the gimbal. The yaw gimbal position scale is graduated in increments of one-half degree from -5 degrees to +13 degrees. Center is located at the +4-degree position due to an inherent offset in the yaw center of gravity. The pitch gimbal position scale is graduated in increments of one-half degree from -9 degrees to +9 degrees, with the center position at 0 degrees.

2.3.4.10 Velocity Change Indicator.

The delta V display provides control and display of spacecraft velocity changes. The display portion of the panel consists of the ΔV REMAINING indicator, which is a five-digit display. The digital display is set by the variable speed ΔV SET switch. Three panel switches control the SPS and RCS engine firing commands. The DIRECT ULLAGE switch is a pushbutton, momentary-contact switch which, when depressed energizes the direct coils of the +X reaction jet solenoids. It serves as a backup for the translation control, which is used under normal conditions for +X translations for SPS fuel settling (ullage). The THRUST ON switch is an illuminated pushbutton, momentary-contact switch which is used to fire the SPS engine in the SCS delta V mode. The switch is enabled when the SPS THRUST switch is in the NORMAL position. The pitch and yaw RCS control and firing circuits are inhibited during SPS engine firing. After ignition, the SPS engine continues to fire until the ΔV REMAINING display counts down to zero. The THRUST ON switch is also used to back up the Apollo guidance computer firing commands in the G&N delta V mode in the event of an AGC failure. The SPS THRUST switch provides overall control of SPS engine firing. The NORMAL position enables normal engine firing sequences. The OFF position provides a positive off command as backup in case of a malfunction. The DIRECT ON position provides a positive firing command to the SPS engine.

The ΔV SET switch sets up the ΔV REMAINING display and logic circuits prior to delta V maneuvers. Pressing the upper section of the ΔV switch sets up logic circuitry which causes a slew motor to drive the ΔV REMAINING display in the positive direction (increasing velocity). Pressing the lower section of the switch drives the display in the negative direction (decreasing velocity). The switch has two sets of contacts in both the upper and lower positions. Pressing the switch lightly engages the first set of contacts, which results in a signal that will drive the display at a rate of two feet per second. Pressing the switch harder (past an easily felt hard/soft point) engages the second set of contacts, which results in the display being driven at a rate of 64 feet per second. As the display is driven to the desired velocity, the integrator and logic circuit is set to a logical value equal to the desired change. The translation command signal energizes a relay which applies the output of the AGAA accelerometer to the integrator and logic circuit, which compares the desired velocity increment with the actual velocity change received from the accelerometer. When the actual velocity change equals the value set, the ΔV REMAINING counter sends a signal to a coincidence detector logic circuit which, in turn, transmits a stop firing command to the thrust control logic. Although this causes the SPS engine to stop firing, some thrusting continues as a result of the tail-off inherent to the engine. The ΔV REMAINING display continues to count past zero (i.e., 99999, 99998, 99997, etc.) until no further acceleration is felt. Tail-off effects will be calculated prior to flight and compensated for by the crew when the display is being set up for delta V maneuvers.

2.3.4.11 Electrical Power Distribution.

See figures 2.3-14 and 2.3-15.

2.3.5 PERFORMANCE AND DESIGN DATA.

Figure 2.3-16 contains the latest available power consumption data for the Block I SCS. The translation controls were not included in the SCS component breakdown because they do not dissipate a significant amount of power within themselves.

The X-axis accelerometer is a servo-rebalanced pendulous accelerometer. Some of the accelerometer design characteristics are as follows.

Operating range	0.001 g to 2 g
Threshold	1×10^{-4} g (max)
Null torque	1×10^{-4} g (max)
Signal generator sensitivity	50 millivolts/millirad
Signal generator null	1 mr (max)

The body-mounted attitude gyros are single-degree-of-freedom gyros. Some of the gyro design characteristics are as follows.

Acceleration sensitive drift	4 deg/hr/g
Random drift	0.05 deg/hr
Maximum torquing rate	25 deg/sec
Maximum input attitude	±20 deg
Gyro threshold	0.02 deg/hr
Gyro transfer function	1.15 millivolts/millirad
Signal generator null	3.5 millivolts
Excitation frequency	400±0.01% cps

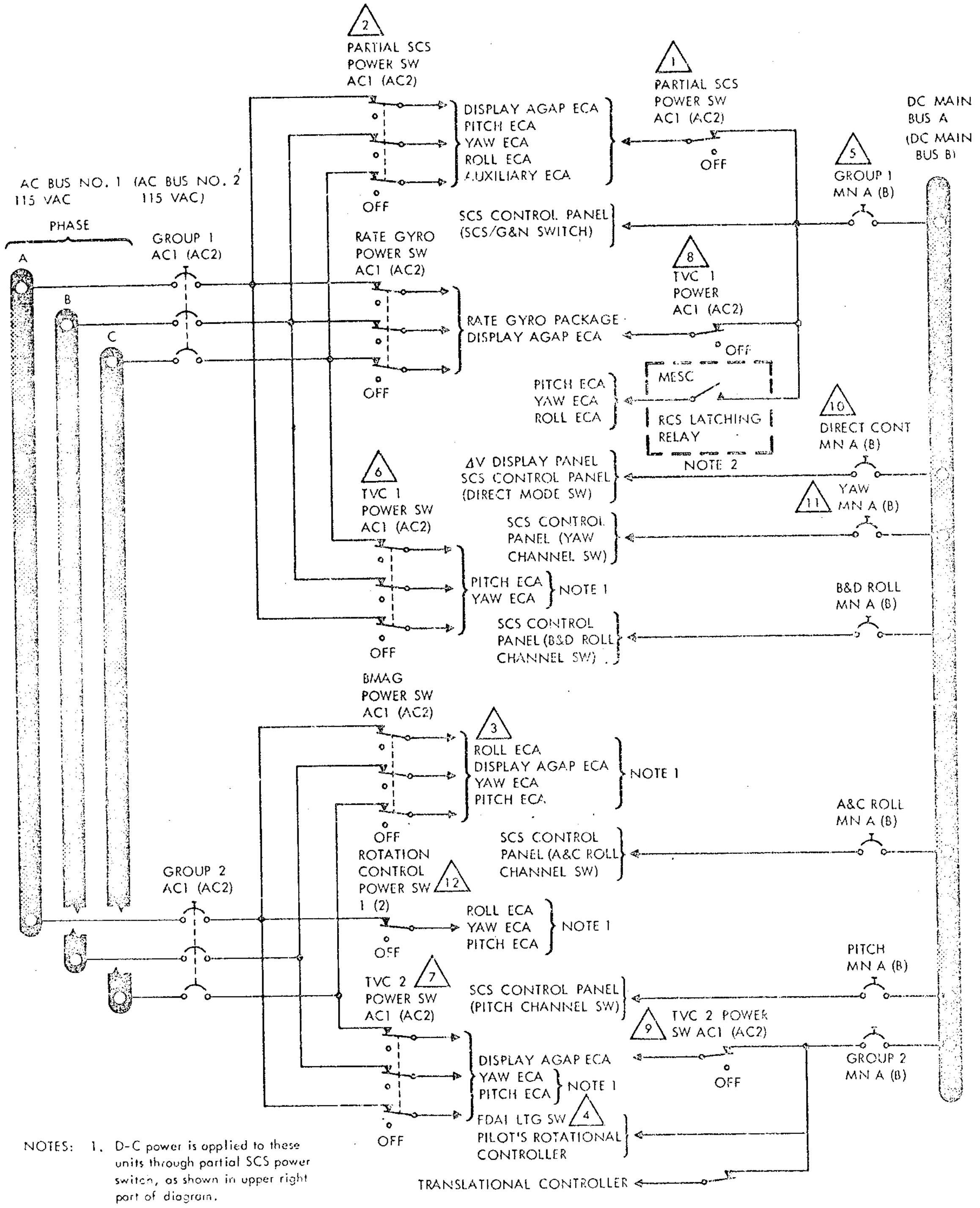
The rate gyros are single-axis miniature gyros. Some of the gyro design characteristics are as follows.

Full-scale range	30°/sec
Input range (to limit stop)	30°/sec
Maximum rate without damage	600°/sec
Excitation frequency	400±0.01% cps
Threshold	0.02°/sec

2.3.6 OPERATIONAL LIMITATIONS AND RESTRICTIONS.

2.3.6.1 S/C Attitude Change.

It is recommended that vehicle attitude change be held to maximum rates of less than 20 degrees per second roll and 5 degrees per second in pitch and yaw to prevent possible loss of attitude reference. Because of the AGCU digital logic, the stepping motor can be stepped at a maximum



NOTES:
1. D-C power is applied to these units through partial SCS power switch, as shown in upper right part of diagram.
2. Contacts close after S-IVB separation or high-altitude abort.

Figure 2.3-14. SCS Power Distribution

SYSTEM CAPABILITIES	REQUIRED SCS POWER SWITCHES					
	PARTIAL SCS POWER	RATE GYRO POWER	BMAG POWER	ROTATION CONTROL POWER	TVC 1 POWER	TVC 2 POWER
BACKUP RATE	●		●			
CHANNEL ENABLE						
DIRECT ROTATION CONTROL						
PROPORTIONAL ROTATION CONTROL	●	●	● (1)	●		
ATTITUDE IMPULSE	●					
G&N SYNC						
TRANSLATION	●					
ΔV SET					●	
SPS ENGINE IGNITION	● (2)					
MANUAL SPS GIMBAL TRIM	●				●	● (3)
DIRECT ULLAGE						
THRUST VECTOR CONTROL	●	●	● (4)		●	● (3)
MANUAL THRUST VECTOR CONTROL			●	●		●
ATTITUDE HOLD	●	● (7)	● (5)			
RATE DAMPING	●	●	● (1)			
FDAI ALIGN	●					
RATE INDICATORS VALID	●	●	● (1)			
BALL & ROLL BUG DRIVEN	●					
ATTITUDE ERROR INDICATORS VALID	● (8)		● (6)			

NOTES:
1. If backup rate has been selected in any axis. If backup rate has been selected in all 3 axes, RATE GYRO POWER need not be on.
2. PARTIAL SCS POWER must be on if ΔV switch is at NORMAL. However, SPS engine can be turned on when all power switches are off, by placing ΔV switch to DIRECT ON.
3. TVC 2 POWER switch must be on if manual TVC is engaged.
4. SCS ΔV mode only. However, in G&N ΔV mode, BMAG POWER must be on if backup rate has been selected in any axis. If backup rate has been selected in all three axes, RATE GYRO POWER switch is set to OFF.
5. SCS modes only; also G&N modes if backup rate has been selected in any axis.
6. SCS modes only, if backup rate has not been selected.
7. RATE GYRO POWER switch may be set to OFF for extended periods of attitude hold.
8. FDAI ALIGN and G&N modes only.

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Figure 2.3-15. SCS Power vs System Capability

SYSTEM STEADY-STATE POWER CONSUMPTION-WATTS (SQ-1)

MODE / SWITCH	MONITOR 115 VAC 400 CPS				G&N ATTITUDE 115 VAC 400 CPS				SCS ATTITUDE 115 VAC 400 CPS				G&N ENTRY 115 VAC 400 CPS			
	φA	φB	φC	DC	φA	φB	φC	DC	φA	φB	φC	DC	φA	φB	φC	DC
PARTIAL SCS PWR (1)	25.2	24.1	17.6	16.7	26.0	24.2	18.3	9.0	27.6	24.3	18.5	11.8	25.1	24.5	18.0	19.8
POWER FACTOR	0.94	0.89	0.91		0.95	0.89	0.88		0.94	0.89	0.88		0.94	0.89	0.88	
FDAI	3.4				3.4				*3.4				3.4			
POWER FACTOR	1.0				1.0				1.0				1.0			
RGA POWER	5.2	3.4	11.8		5.2	3.4	11.8		*5.2	*3.4	*11.8		5.2	3.4	11.8	
POWER FACTOR	0.98	0.85	0.99		0.98	0.85	0.99		*0.98	*0.85	*0.99		0.98	0.85	0.99	
ROT. CONT POWER	5.6				5.6				*5.6				5.6			
POWER FACTOR	0.92				0.92				*0.92				0.92			
TVC SW NO. 1	16.4	8.7	8.7	16.3	16.4	8.7	8.7	16.3	16.4	8.7	8.7	16.3	16.4	8.7	8.7	16.3
POWER FACTOR	0.96	0.86	0.86		0.96	0.86	0.86		0.96	0.86	0.86		0.96	0.86	0.86	
TVC SW NO. 2	11.5	4.3	4.2		11.5	4.3	4.2		11.5	4.3	4.2		11.5	4.3	4.2	
POWER FACTOR	0.75	0.57	0.63		0.75	0.57	0.63		0.75	0.57	0.63		0.75	0.57	0.63	
BMAG POWER	6.1	3.4	7.0	62.3	6.1	3.4	7.0	62.3	*6.1	*3.4	*7.0	*62.3	6.1	3.4	7.0	62.3
POWER FACTOR	0.99	0.78	0.80		0.99	0.78	0.80		*0.99	*0.78	*0.80		0.99	0.78	0.80	
CONT PANEL INPUT (1)				11.82				5.13				4.22				28.13

SYSTEM STEADY-STATE POWER CONSUMPTION-WATTS (SQ-1) (CONTINUED)

MODE / SWITCH	SCS ENTRY 115 VAC 400 CPS				G&N ΔV 115 VAC 400 CPS				SCS ΔV 115 VAC 400 CPS				LOCAL VERTICAL 115 VAC 400 CPS			
	φA	φB	φC	DC	φA	φB	φC	DC	φA	φB	φC	DC	φA	φB	φC	DC
PARTIAL SCS PWR (1)	27.5	24.5	18.3	22.9	25.7	24.0	18.6	59.4	28.3	23.3	19.0	61.3	26.1	22.6	17.6	12.4
POWER FACTOR	0.94	0.89	0.88		0.94	0.88	0.91		0.93	0.86	0.91		0.94	0.88	0.90	
FDAI	3.4				3.4				3.4				3.4			
POWER FACTOR	1.0				1.0				1.0				1.0			
RGA POWER	5.2	3.4	11.8		5.2	3.4	11.8		5.2	3.4	11.8		5.2	3.4	11.8	
POWER FACTOR	0.98	0.85	0.99		0.98	0.85	0.99		0.98	0.85	0.99		0.98	0.85	0.99	
ROT. CONT POWER	5.6				5.6				5.6				5.6			
POWER FACTOR	0.92				0.92				0.92				0.92			
TVC SW NO. 1	16.4	8.7	8.7	16.3	16.4	8.7	8.7	16.3	*16.4	*8.7	*8.7	*16.3	16.4	8.7	8.7	16.3
POWER FACTOR	0.96	0.86	0.86		0.96	0.86	0.86		*0.96	*0.86	*0.86		0.96	0.86	0.86	
TVC SW NO. 2	11.5	4.3	4.2		*11.5	*4.3	*4.2		11.5	4.3	4.2		11.5	4.3	4.2	
POWER FACTOR	0.75	0.57	0.63		*0.75	*0.57	*0.63		0.75	0.57	0.63		0.75	0.57	0.63	
BMAG POWER	6.1	3.4	7.0	62.3	6.1	3.4	7.0	62.3	6.1	3.4	7.0	62.3	6.1	3.4	7.0	62.3
POWER FACTOR	0.99	0.78	0.80		0.99	0.78	0.80		0.99	0.78	0.80		0.99	0.78	0.80	
CONT PANEL INPUT (1)				23.07				13.33				6.95				5.65

NOTES:

1. Power measurements made in each mode.
*2. Power consumption assumed same in each mode. Asterisk indicates mode in which measurement was made.
3. Measurements made with SPS engine off. (Two solenoids and 2 relays.)
4. Line losses and inverter efficiencies not considered.
5. All values indicated are in watts.
6. Figure does not include energizing RCS engines for maneuvers.

SM-2A-754D

Figure 2.3-16. SCS Steady-State Power Consumption Data rate of 100 steps per second in roll and 25 steps per second in pitch and yaw. This is equivalent to gyro torquing rates of 20 degrees per second in roll and 5 degrees per second in pitch and yaw. Therefore, if vehicle rates exceed the gyro torquing rates, the stepping motor will lag behind the vehicle rates. Under this condition, the BMAG-AGCU loop may not be able to maintain the SCS attitude reference.

2.3.7 TELEMETRY MEASUREMENTS.

The following is a complete listing of all SCS telemetry data that is monitored by flight controllers and ground support personnel. The last column contains the name and type of S/C crew display. The display utilizes the same pickup or signal source as telemetry, unless a separate measurement number is included in the display column.

An asterisk (*) by the measurement number denotes information which is not available for recording or telemetry transmission during PCM low-bit rate operation.

Figure 2.3-4, sheet 2, identifies telemetry measurements for the SCS yaw channel. Identical measurements in the following list, relative to the roll and pitch channels, are taken from the same circuit junctions as for the yaw channel.

Measurement Number	Description	Sensor Range	Crew Display
* CH 0024 V	Pitch rate		FDAI
* CH 0025 V	Pitch rate manual rotation control	Zero/+5 VRMS	FDAI
* CH 0034 V	Pitch pos feedback in	-6/+6 vdc	None
* CH 0047 V	PTV diff clutch volts comb	TBD	None
CH 0050 V	Pitch rate error amp out	-6.2 to +6.2 vdc	FDAI
CH 0067 V	Pitch integrator/att error summing	TBD	None
* CH 0074 V	MTVC pitch rate	TBD	None
CH 0075 V	Pitch SCS att error	Zero/+10 VRMS	FDAI
* CH 0087 X	+ Pitch/+X solenoid driver out	Off/on event	None
* CH 0088 X	- Pitch/+X solenoid driver out	Off/on event	None
* CH 0089 X	+ Pitch/-X solenoid driver out	Off/on event	None
* CH 0090 X	- Pitch/-X solenoid driver out	Off/on event	None
CH 0100 X	G-N DV mode control	Event	None
CH 0101 X	G-N DV att mode control	Off/on event	None
CH 0102 X	G-N entry mode control	Off/on event	None
CH 0103 X	Monitor mode control	Off/on event	None
* CH 1024 V	Yaw rate	-30 to +30°/sec	FDAI
* CH 1025 V	Yaw manual rotation control	Zero/+5 VRMS	FDAI
* CH 1034 H	Yaw pos feedback in	-8.5/+8.5 vdc	None
* CH 1047 V	YTV diff clutch volts comb	-85 to +85 MADC	None
CH 1050 V	Yaw rate error amp out	-6.2 to +6.2 vdc	FDAI
CH 1067 V	Y integrator/att error summing	-2.5 to +2.5 vdc	None
* CH 1074 V	MTVC yaw rate	TBD	None

Measurement Number	Description	Sensor Range	Crew Display
CH 1075 V	V yaw SCS att error	Zero/+10 VRMS	FDAI
* CH 1087 X	+ Yaw/+X solenoid driver out	Off/on event	None
* CH 1088 X	- Yaw/+X solenoid driver out	Off/on event	None
* CH 1089 X	+ Yaw/-X solenoid driver out	Off/on event	None
* CH 1090 X	- Yaw/-X solenoid driver out	Off/on event	None
CH 1100 X	SCS DV mode control	Event	None
CH 1101 X	SCS att mode control	Event	None
CH 1102 X	SCS entry mode control	Event	None
CH 1103 X	SCS local vertical mode control	Event	None
CH 1104 X	MTVC mode control	Event	None
* CH 2015 V	Combined AG SMRD	0 to 5 vdc	None
* CH 2024 V	Roll rate	TBD	FDAI
* CH 2025 V	Roll man rotation control out	Zero/+5 VRMS	FDAI
* CH 2026 V	Combined RG SMRD	-0 to +5 vdc	None
* CH 2030 T	Combined attitude gyro temp	0 to +5 vdc	AGAP TEMP LIGHT
CH 2050 V	Roll rate error amp out	-6.2 to +6.2 vdc	FDAI
* CH 2070 V	Roll attitude error amp out	-25/+25 vdc	None
CH 2075 V	Roll SCS att error	Zero/10 VRMS	FDAI
* CH 2087 X	+ Roll/+Z solenoid driver out	Event	None
* Ch 2088 X	- Roll/-Z solenoid driver out	Event	None
* Ch 2089 X	+ Roll/-Z solenoid driver out	Event	None
* CH 2090 X	- Roll/-Z solenoid driver out	Event	None
* CH 2091 X	+ Roll/+Y solenoid driver out	Event	None
* CH 2092 X	- Roll/+Y solenoid driver out	Event	None
* CH 2093 X	+ Roll/-Y solenoid driver out	Event	None
* CH 2094 X	- Roll/-Y solenoid driver out	Event	None
CH 3185 X	.05 g Manual switch	Event	None
* CH 3186 V	DV remaining pot out	-3 to +13 KFPS	ΔV Counter
CH 4100 H	Resolver sin out pitch att	-12 to +12 VRMS	None
CH 4101 H	Resolver cos out pitch att	-12 to +12 VRMS	None
CH 4102 H	Resolver sin out yaw att	-12 to +12 VRMS	None
CH 4103 H	Resolver cos out yaw att	-12 to +12 VRMS	None
CH 4104 H	Resolver sin out roll att	-12 to +12 VRMS	None
CH 4105 H	Resolver cos out roll att	-12 to +12 VRMS	None
* CH 4320 X	SPS solenoid driver out 1	Event	None
* CH 4321 X	SPS solenoid driver out 2	Event	None
CG 0001 V	Computer digital data 40 bits	Event	Delta V Display

SECTION 2

SUBSECTION 2.4

SERVICE PROPULSION SYSTEM (SPS)

2.4.1 FUNCTIONAL DESCRIPTION.

The service propulsion subsystem provides the major impulse for all major velocity changes (ΔV's) throughout a mission, plus the S/M abort after the launch escape tower is jettisoned. To perform this function, the susbystem incorporates a helium pressurization system, a propellant feed system, a propellant gauging and utilization system, and a rocket engine. The oxidizer is nitrogen tetroxide, and the fuel is a blended hydrazine (approximately 50% unsymmetrical dimethyl hydrazine and 50% anhydrous hydrazine). The pressurizing gas is helium. The subsystem incorporates displays and sensing devices to permit ground-based stations and the crew to monitor its operation.

A functional flow diagram is contained in figure 2.4-1.

The helium pressure is directed to the helium pressurizing valves which isolate the helium during nonthrusting periods, or allows the helium to pressurize the fuel and oxidizer tanks during thrusting periods. The helium pressure is reduced at the pressure regulators to a desired working pressure. The regulated helium pressure is directed through check valves that permit helium flow in the downstream direction when the pressurizing valves are open, and prevent a reverse flow of propellants during non-thrusting periods. The heat exchangers transfer heat from the propellants to the helium gas to reduce any pressure excursions that may result from a temperature differential between the helium gas and propellants in the tanks. The relief valves maintain the structural integrity of the propellant tank systems if an excessive pressure rise occurs.

The total propellant supply is contained within four similar tanks: an oxidizer storage tank, oxidizer sump tank, fuel storage tank, and fuel sump tanks. The storage and sump tanks for each propellant system are connected in series by a single transfer line. The regulated helium enters into the fuel and oxidizer storage tank, pressurizing the storage tank propellants, and forces the propellant to an outlet in the storage tank which is directed through a transfer line into the respective sump tank standpipe, pressurizing the propellants in the sump tank. The propellant in the sump tank is directed to the exit end into a propellant retention reservoir, retaining sufficient propellants at the tank outlets to permit engine restart capability in a zero g condition. The propellants exit from the respective sump tanks into a single line to the heat exchanger.

A propellant utilization valve is installed in the oxidizer line, and is powered only during SPS thrusting periods. The propellant utilization valve aids in maintaining the center of mass in a relatively confined area in the Y - Z plane, in addition to simultaneous propellant depletion. The oxidizer supply and fuel supply are connected from the sump tank to the engine interface flange.

The propellants flow from the propellant sump tank feed through their respective plumbing to the main propellant orifices and filters to the bipropellant valve. The bipropellant valve assembly contains the main propellant valves that are pneumatically controlled and allows the propellant valves to distribute the propellants to the engine injector during thrusting periods, or isolates the propellants from the injector during nonthrusting periods.

The engine injector distributes the propellants through orifices in the face of the injector where the fuel and oxidizer impinge upon each other, atomize, and ignite due to hypergolic ignition.

The ablative combustion chamber absorbs the heat generated within the chamber. The nozzle extension is attached to the ablative chamber and radiates to space.

The engine assembly is mounted to the structure of the service module and is gimbaled to permit thrust vector alignment through the center of mass, prior to thrust initiation and thrust vector control during a thrusting period. A flight combustion stability monitor system is employed to monitor the engine for instability during thrusting periods.

Propellant quantity is measured by two separate sensing systems: primary and auxiliary. The sensing systems are powered only during thrust-on periods due to the capacitance and point sensor measuring techniques. The linearity would not provide accurate indications during the zero g SPS nonthrusting periods.

The control of the subsystem is automatic with provisions for manual override.

2.4.2 MAJOR COMPONENT/SUBSYSTEM DESCRIPTION.

2.4.2.1 Pressurization Subsystem.

The pressurization subsystem consists of two helium tanks, two helium pressurizing valves, two dual pressure regulator assemblies, two dual check valve assemblies, two pressure relief valves, and two heat exchangers. The critical components are redundant to increase reliability.

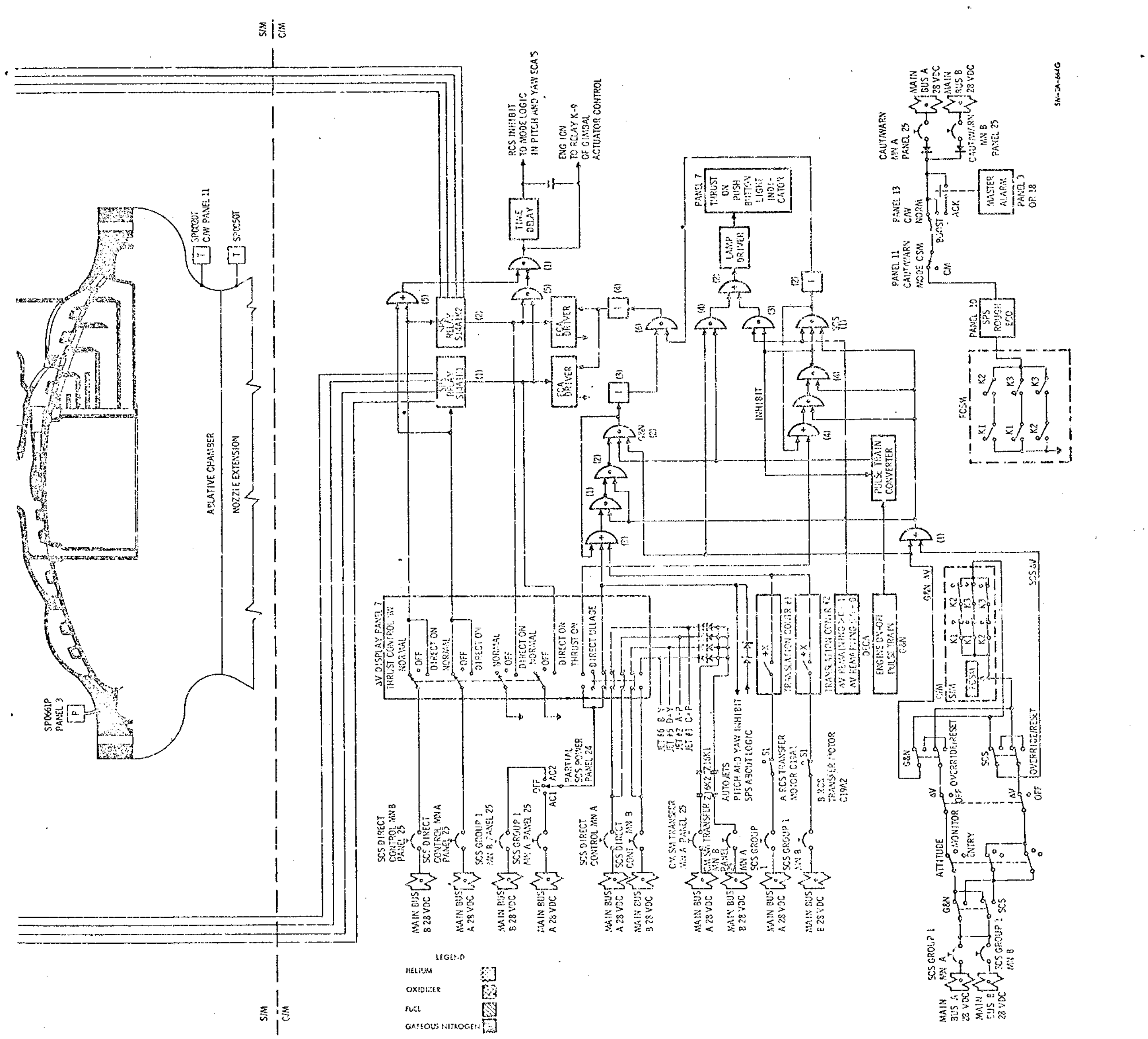
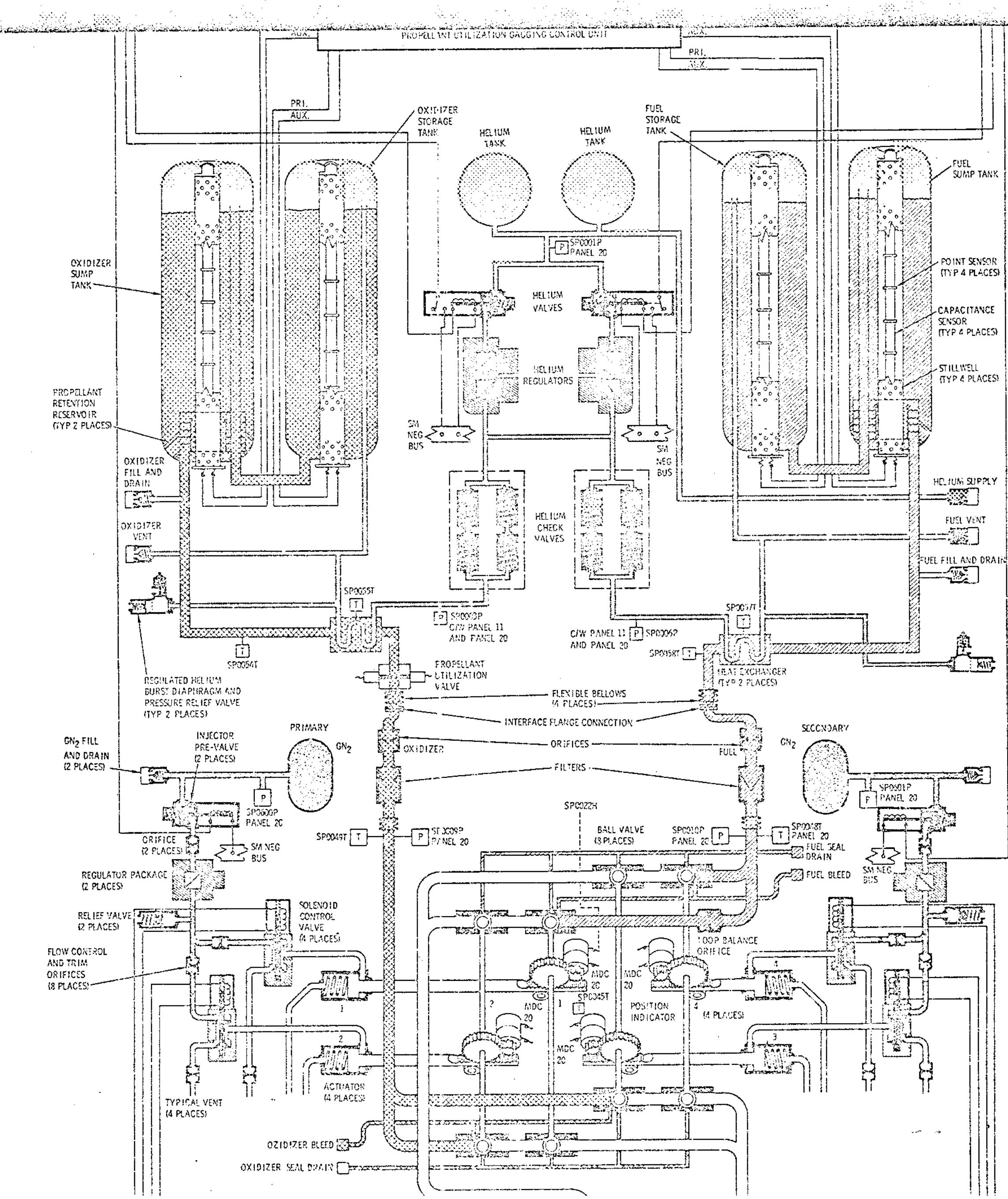
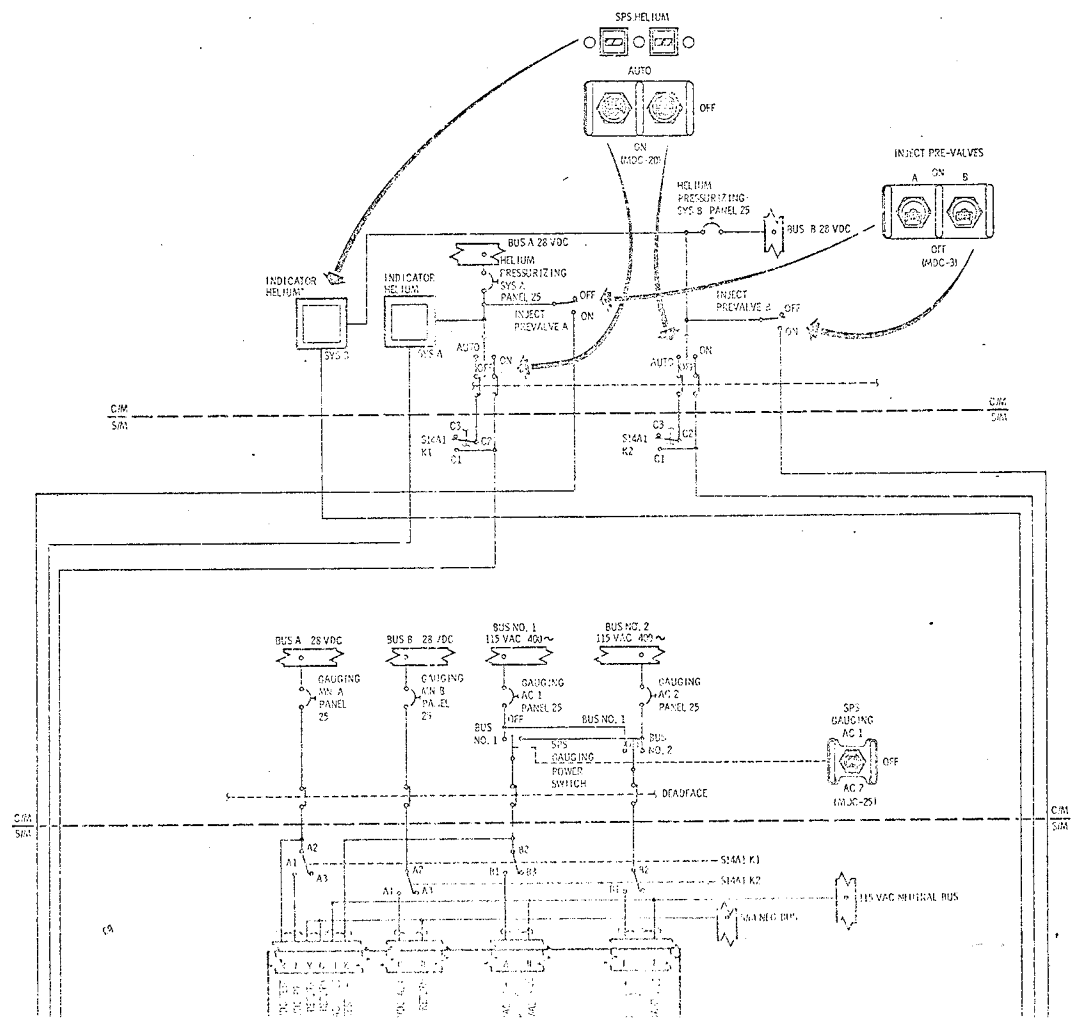


Figure 2.4-1. SPS Functional Flow Diagram



PROPELLANT UTILIZATION GAGGING CONTROL UNIT
AUX.
PRI.
AUX.
PRI.
AUX.
OXIDIZER STORAGE TANK
HELIUM TANK
HELIUM TANK
FUEL STORAGE TANK
FUEL SUMP TANK
OXIDIZER SUMP TANK
SP0001P PANEL 20
HELIUM VALVES
POINT SENSOR (TYP 4 PLACES)
CAPACITANCE SENSOR (TYP 4 PLACES)
HELIUM REGULATORS
STILLWELL (TYP 4 PLACES)
PROPELLANT RETENTION RESERVOIR (TYP 2 PLACES)
SM NEG BUS
SM NEG BUS
OXIDIZER FILL AND DRAIN
HELIUM SUPPLY
HELIUM CHECK VALVES
FUEL VENT
OXIDIZER VENT
FUEL FILL AND DRAIN
SP0055T
SP0017T
SP0003P C/W PANEL 11 AND PANEL 20
C/W PANEL 11 AND PANEL 20
SP0006P
SP0054T
SP0015RT
PROPELLANT UTILIZATION VALVE
HEAT EXCHANGER (TYP 2 PLACES)
REGULATED HELIUM BURST DIAPHRAGM AND PRESSURE RELIEF VALVE (TYP 2 PLACES)
FLEXIBLE BELLOWS (4 PLACES)
INTERFACE FLANGE CONNECTION
PRIMARY
SECONDARY
INJECTOR PRE-VALVE (2 PLACES)
ORIFICES
OXIDIZER
FUEL
FILTERS
SP0901P PANEL 20
SP0900P PANEL 20
SP0022H
SP0049T
SP0009P PANEL 20
BALL VALVE (8 PLACES)
SP0010P PANEL 20
SP0048T PANEL 20
ORIFICE (2 PLACES)
SM NEG BUS
FUEL SEAL DRAIN
FUEL BLEED
SM NEG BUS
REGULATOR PACKAGE (2 PLACES)
SOLENOID CONTROL VALVE (4 PLACES)
RELIEF VALVE (2 PLACES)
LOOP BALANCE ORIFICE
FLOW CONTROL AND TRIM ORIFICES (8 PLACES)
MDC 20
MDC 20
SP0045T
POSITION INDICATOR
(4 PLACES)
MDC 20
MDC 20
ACTUATOR (4 PLACES)
TYPICAL VENT (4 PLACES)
OXIDIZER BLEED
OXIDIZER SEAL DRAIN



SPS HELIUM
AUTO
OFF
ON
(MDC-20)
INJECT PRE-VALVES
A
ON
B
OFF
(MDC-3)
HELIUM PRESSURIZING SYS B PANEL 25
BUS B 28 VDC
BUS A 28 VDC
HELIUM PRESSURIZING SYS A PANEL 25
INDICATOR HELIUM
SYS B
INDICATOR HELIUM
SYS A
INJECT PREVALVE A
INJECT PREVALVE A
AUTO
C/M
S/M
S/M
C/M
C3
C2
C1
S14A1 K1
S14A1 K2
BUS A 28 VDC
BUS B 28 VDC
BUS NO. 1 115 VAC 400~
BUS NO. 2 115 VAC 400~
GAUGING MN A PANEL 25
GAUGING MN B PANEL 25
GAUGING AC 1 PANEL 25
GAUGING AC 2 PANEL 25
SPS GAUGING AC 1
OFF
AC 2
(MDC-25)
BUS NO. 1
BUS NO. 2
SPS GAUGING POWER SWITCH
DEADFACE
A1
A2
A3
B1
B2
B3
S14A1 K1
S14A1 K2
115 VAC NEUTRAL BUS

2.4.2.1.1 Helium Tanks.

The two helium supply spherical pressure vessels are located in the center section of the service module.

2.4.2.1.2 Helium Pressurizing Valves.

These two continuous duty solenoid-operated valves are normally closed to the helium supply pressure. The valves are energized open and spring-loaded closed. When the helium switches on panel 20 are in the AUTO position, the valves are energized and de-energized by the thrust ON-OFF signal. The valves may be manually controlled by the crew, utilizing the ON and OFF position of the helium switches. Each valve is controlled individually by a separate switch on panel 20.

Each valve contains a position switch which controls a position indicator above each switch on panel 20. When a valve is closed, the position switch is open and the indicator is gray (same color as the panel). When a valve is open, the position switch is closed and the indicator has diagonal lines, indicating the valve is open.

In the event of a failure in the automatic mode of operation, the crew manually enables the pressurizing valve or valves on panel 20. In the event of a problem with one of the regulating units, the crew manually isolates the required pressurizing valve.

2.4.2.1.3 Pressure Regulator Assemblies.

Pressure regulation is accomplished by two pressure regulating units in parallel, downstream of each helium pressurizing valve. Each regulator unit contains a primary and secondary regulator in series, and a pressure surge damper installed on the inlet to each regulating unit.

The primary regulator is normally the controlling regulator, while the secondary is normally open during a dynamic flow condition. The secondary regulator will not become a controlling regulator until the primary, due to a problem, allows a higher pressure than normal, and allows the secondary regulator to function and become the controlling regulator. All regulator pressures are in reference to a bellows assembly that is vented to ambient.

One of the regulating units incorporated in one of the parallel paths is considered the working regulator. The regulating unit in the remaining parallel path is normally locked up when the system is dynamic. The regulator that is normally locked up would not function until the normal regulator allowed the regulated helium pressure to decrease, due to a problem, and allow the normally locked-up regulator to become the functioning regulator.

2.4.2.1.4 Check Valve Assemblies.

Each assembly contains four independent check valves connected in a series-parallel configuration which provides added redundancy. The check valves will not open until there is a 4-psi pressure differential between the check valve assembly inlet and outlet lines.

2.4.2.1.5 Helium Pressure Relief Valves.

The pressure relief valves consist of a relief valve, a burst diaphragm, and a filter.

In this event, excessive helium and/or propellant vapor ruptures the burst diaphragm and the filter (filter is to be a nonfragmentation type) filters any fragmentation. The relief valve opens and vents the applicable system to space, and will close and reseal after the excessive pressure has returned to the operating level. The burst diaphragm provides a more positive seal of helium than a relief valve. The filter prevents any fragments from the diaphragm from entering the relief valve seat.

A pressure bleed device is incorporated between the burst diaphragm and relief valve. The bleed valve vents the cavity between the burst diaphragm and relief valve in the event of any leakage from the diaphragm. The bleed device is normally open and will close when the pressure increases up to a predetermined pressure.

2.4.2.1.6 Heat Exchangers.

Each unit is a line-mounted, counterflow heat exchanger consisting of a portion of the helium pressurization line, coiled helically within an enlarged section of the propellant supply line. This arrangement causes the helium gas, flowing through the coiled line, to approach the temperature of the propellant.

2.4.2.2 <u>Propellant Subsystem</u>.

This subsystem consists of two fuel tanks (storage and sump), two oxidizer tanks (storage and sump), propellant feed lines, and bipropellant valve assembly.

2.4.2.2.1 Propellant Tanks.

The total propellant supply is contained within four hemispherical-domed cylindrical tanks within the service module. The storage tanks are pressurized from the helium supply, and an outlet transfers the propellant and/or helium gas from the storage tanks through their respective transfer lines to the sump tanks. A standpipe, in the sump tanks, allows the propellant and/or helium gas from the storage tanks to pressurize the sump tanks. Thus, the propellants in the sump tanks are directed in the retention reservoirs, to the outlet, and then to the engine.

The retention reservoirs are installed in the exit end of the sump tanks only. The umbrella propellant retention screens retain a quantity of propellant in the exit end of the sump tanks, in addition to retaining propellant within the retention reservoir can and in the plumbing from the sump tank outlet to the bipropellant valve of the SPS engine during zero g conditions. An ullage maneuver must be performed prior to a thrust-on signal to the SPS engine by the S/M RCS. The ullage maneuver is performed to ensure that no gases are retained below or aft of the retention screens, in addition to settling the propellants and ensuring that the retention screens are not damaged.

2.4.2.2.2 Tank Propellant Feed Lines.

The propellant feed lines have flexible bellows assemblies installed to permit alignment of the tank feed plumbing to the engine interface plumbing.

2.4.2.3 Bipropellant Valve Assembly.

The bipropellant valve assembly consists of two gaseous nitrogen (GN_2) pressure vessels, two injector prevalves, two GN_2 regulators, two GN_2 relief valves, four solenoid control valves, four actuators, and eight propellant ball valves.

2.4.2.3.1 Gaseous Nitrogen (GN_2) Pressure Vessels.

Two GN_2 tanks are mounted on the bipropellant valve assembly that supply pressure to the injector prevalves. One GN_2 tank is in the primary pneumatic control system A and the remaining GN_2 tank is in the secondary pneumatic control system B.

2.4.2.3.2 Injector Prevalves.

The injector prevalves are two-position solenoid-operated valves, one for each pneumatic control system and identified as A and B. The valve is energized open and spring-loaded closed. The injector prevalve is opened upon command from the crew, panel 3, prior to an engine thrusting period. The injector prevalves energized open allows GN_2 supply tank pressure to flow to an orifice and on into the regulator.

2.4.2.3.3 GN_2 Pressure Regulators.

A single-stage regulator is installed in each pneumatic control system between the injector prevalves and the solenoid control valves. The regulator reduces the supply GN_2 pressure to a desired working pressure.

2.4.2.3.4 GN_2 Relief Valves.

A pressure relief valve is installed in each pneumatic control system, downstream of the GN_2 pressure regulators, to limit the pressure applied to the solenoid control valves in the event a GN_2 pressure regulator malfunctioned open. The orifice, between the injector prevalve and regulator, is installed to restrict the flow and allow the relief valve to relieve the pressure in event the regulator fails to open, preventing any damage to the solenoid control valves and/or actuators.

2.4.2.3.5 GN_2 Solenoid Control Valves.

Four solenoid-operated, three-way, two-position control valves are utilized for actuator control. Two solenoid control valves are located downstream of the GN_2 regulators in each pneumatic control system. The two solenoid control valves in the primary system are identified in 1 and 2, and the two in the secondary system are identified as 3 and 4. The two solenoid control valves in the primary system control the actuator and ball valves 1 and 2. The two solenoid control valves in the secondary system control the actuator and ball valves 3 and 4. The SPS thrust ON-OFF command controls the energizing or de-energizing of all four solenoid control valves.

2.4.2.3.6 GN_2 Ball Valve Actuators.

Four piston-type, pneumatically operated actuators are utilized to control the eight propellant ball valves. Each actuator piston is mechanically connected to a pair of propellant ball valves; one fuel and one oxidizer. When the solenoid control valves are opened, pneumatic pressure is applied to the opening side of the actuators. The spring pressure, on the closing side, is overcome and the actuator piston moves. Utilizing a rack and pinion gear, linear motion of the actuator connecting arm is converted into rotary motion, which opens the propellant ball valves. When the engine firing signal is removed from the solenoid control valves, the solenoid control valves close removing pneumatic pressure source from the opening side of the actuators. The actuator closing side spring pressure now forces the actuator piston to move in the opposite direction, causing the propellant ball valves to close. The piston movement forces the remaining GN_2, on the opening side of the actuator, back through the solenoid control valves where they are vented overboard.

2.4.2.3.7 Bipropellant Valves.

The eight propellant ball valves are used to distribute fuel and oxidizer to the engine injector assembly. Four linked pairs, each pair consisting of one fuel and one oxidizer ball valve controlled by a single actuator, are arranged in a series-parallel configuration. The parallel arrangement provides redundancy to ensure engine ignition, and the series arrangement to ensure thrust termination. When the actuators are opened, each propellant ball valve is rotated, thereby aligning the ball to a position that allows propellants to flow to the engine injector assembly. The mechanical arrangement is such that the oxidizer ball valves maintain an 8-degree lead over the fuel ball valves upon opening, which results in smoother engine starting transients. Each pair of propellant valves incorporates a potentiometer drive gear and housing. Each housing contains two potentiometers that are mechanically driven: one provides an electrical signal converted into a visual indication of valve position of panel 20, and the remaining provides a signal to telemetry.

2.4.2.3.8 Engine Propellant Lines.

Integral propellant lines are utilized, on the engine, to route each propellant from the interface points in the gimbal plane area to the propellant engine gimbaling, orifices for adjustment of oxidizer/fuel ratio, and screens for keeping particle contaminants from entering the engine.

2.4.2.4 Engine Injector.

The injector is bolted to the ablative thrust chamber attach pad. Propellant distribution to the injector orifices, on the combustion face of the injector, is accomplished through concentric annuli machined in the face of the injector assembly and covered by concentric closeout rings. Propellant distribution to the annuli is accomplished through alternate radial manifolds welded to the backside of the injector body. The injector is regeneratively cooled and baffled to provide combustion stability. The fuel and oxidizer orifices impinge upon each other, atomize, and ignite due to hypergolic reaction.

2.4.2.5 Ablative Combustion Chamber.

The ablative combustion chamber material extends from the injector attach pad to the nozzle extension attach pad. The ablative material consists of a liner, a layer of insulation, integral metal attach flanges for mounting the injector and nozzle extension, and structural outer reinforcement. The only restriction on a restart is dependent upon the SPS WALL TEMP HI caution and warning light, on panel 11, monitoring the outside wall temperature at the throat.

2.4.2.6 Nozzle Extension.

The bell-contoured, nozzle extension is bolted to the ablative thrust chamber exit area. The nozzle extension is radiant-cooled and contains an external stiffener to provide additional strength.

2.4.2.7 Flight Combustion Stability Monitor (FCSM).

The FCSM ia an accelerometer package, mounted to the SPS engine injector, to monitor the engine for vibration buildup characteristic of combustion instability.

The three accelerometers provide signals into an FCSM box assembly which amplifies it and, when the vibration g-level exceeds 180 g's peak to peak for 70±20 milliseconds, a level detector triggers a power switch gating power to the summing logic. The stability monitor will monitor only in a frequency range where instability would occur (approximately 600 to 5000 cycles per second).

The summing logic, if there are two or more rough combustion signals received from the normally closed contacts, will open removing power from the G&N and SCS driver logic; and will close the normally open contacts providing a ground to the SPS ROUGH ECO caution and warning light, on panel 10, informing the crew the SPS engine was shut down due to combustion instability.

The FCSM trigger circuits will provide power to the voting logic relays continuously once unstable combustion is sensed; and power remains applied to the voting logic relays even though the engine is shut down, unstable combustion is no longer sensed, and the SPS ROUGH ECO remains illuminated.

When the engine is shut down to unstable combustion and to remove power from the voting logic relays, the crew would place any one of the following switches to the position indicated to reset the FCSM voting logic:

a. AUTO-OVERRIDE switches to OVERRIDE.
b. ΔV switch to OFF.
c. ATTITUDE-MONITOR-ENTRY switch to either MONITOR or ENTRY.
d. FCSM SCS and G&N switches to RESET/OVERRIDE.

When the FCSM is reset, the SPS ROUGH ECO light on panel 10 will not be illuminated.

The RESET/OVERRIDE switches, on panel 2, provide a bypass capability of the FCSM system. With the FCSM RESET/OVERRIDE switches, on panel 2, in the RESET/OVERRIDE position, power is not provided to the FCSM box and summing logic assemblies rendering the FCSM inoperative as well as the SPS ROUGH ECO caution and warning light on panel 10. If unstable combustion occurs, the engine continues to thrust.

The FCSM box assembly and summing logic assembly receives power from the SCS through the following switches:

a. Mode select switch to G&N or SCS mode, panel 8.
b. ATTITUDE-MONITOR-ENTRY switch to ATTITUDE, panel 8.
c. ΔV switch to ΔV position, panel 8.
d. FCSM G&N, RESET/OVERRIDE switch to G&N position, panel 2.
e. FCSM SCS, RESET/OVERRIDE switch to SCS position, panel 2

2.4.2.8 SPS Electrical Heaters.

Twenty-six electrical strip heaters are employed on the propellant line brackets, fuel and oxidizer heat exchangers, propellant utilization valve, bipropellant valve assembly, and the injector tubes. These electircal strip heaters are employed to control the temperature of the entire aft bulkhead and engine areas, thus the propellants.

The strip heaters are controlled as a normal manual function of the crew, on panel 19 (figure 2.4-2), utilizing the SPS HEATER switch. The crew will place the switch to position A or B when the propellant temperature indicator, on panel 20, reads - 40°F (lower red line indication) which is equivalent to a propellant temperature of +40°F. The propellant temperature readout is derived from the engine fuel-feed line measurement (figure 2.4-1). The crew will place the SPS HEATER switch to OFF when the propellant temperature indicator, on panel 20, reads +80°F (upper red line indication) which is equivalent to a propellant temperature of +120°F.

Propellant Temperature Comparison

Propellant Temperature Indicator Reading °F															
-100	-80	-60	-40	-20	0	+20	+40	+60	+80	+100	+120	+140	+160	+180	+200
Equivalent Temperature °F															
+0	+13.33	+26.67	+40	+53.34	+66.67	+80	+93.34	+106.67	+120	+133.34	+146.67	+160	+173.34	+186.67	+200

When the SPS HEATER switch is placed to position A, all 26 heating elements (A) will receive 28 vdc. When the SPS HEATER switch is placed to position B, all 26 heating elements (B) will receive 28 vdc. The OFF position of the SPS HEATER switch removes all electrical power from the SPS heating elements.

2.4.2.9 Thrust Mount Assemblies.

The thrust mount assembly consists of a gimbal ring, engine-to-vehicle mounting pads, and gimbal ring-to-combustion chamber assembly supports. The thrust structure is capable of providing ±8.5 degrees inclination about the Z-axis and ±7.5 degrees about the Y-axis.

2.4.2.9.1 Gimbal Actuator.

Thrust vector control of the service propulsion engine is achieved by dual-, servo-, electro-mechanical actuators. The gimbal actuators are capable of providing control around the Z-Z axis (yaw) of 7 (+1/2, -0) degrees in either direction from a +4-degree null offset, and around the Y-Y axis (pitch) of 6(+1/2, -0) degrees in either direction from a zero-degree null offset. The reason for the 4-degree offset to the +Y axis is due to the offset center of mass in the spacecraft.

Each actuator assembly (figure 2.4-3) consists of four electromagnetic particle clutches, two d-c motors, a bull gear, jackscrew and ram, ball nut, four linear position transducers, and two velocity generators. The actuator assembly is a sealed unit and enclose those portions protruding from the main housing.

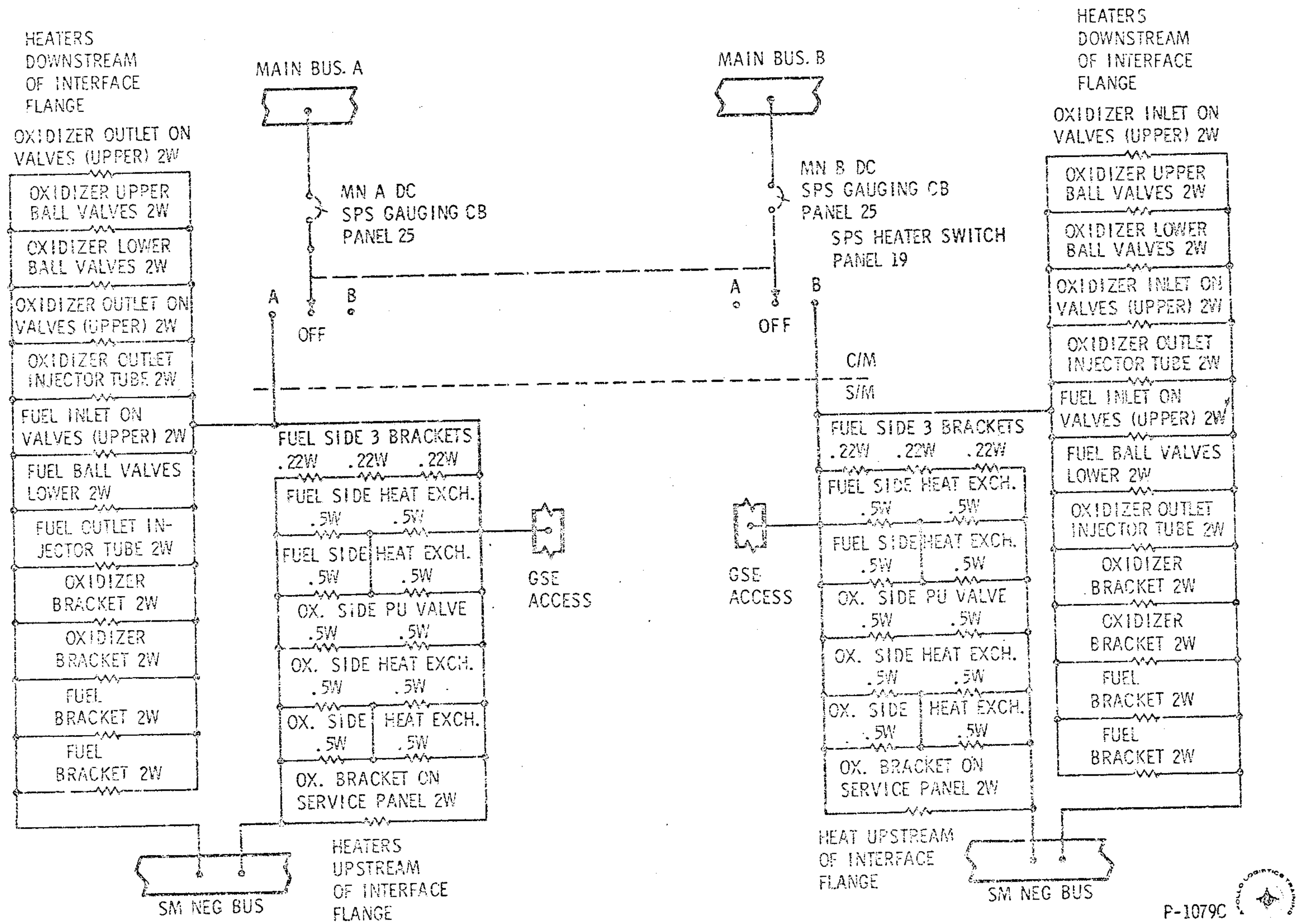


Figure 2.4-2. SPS Electrical Heaters

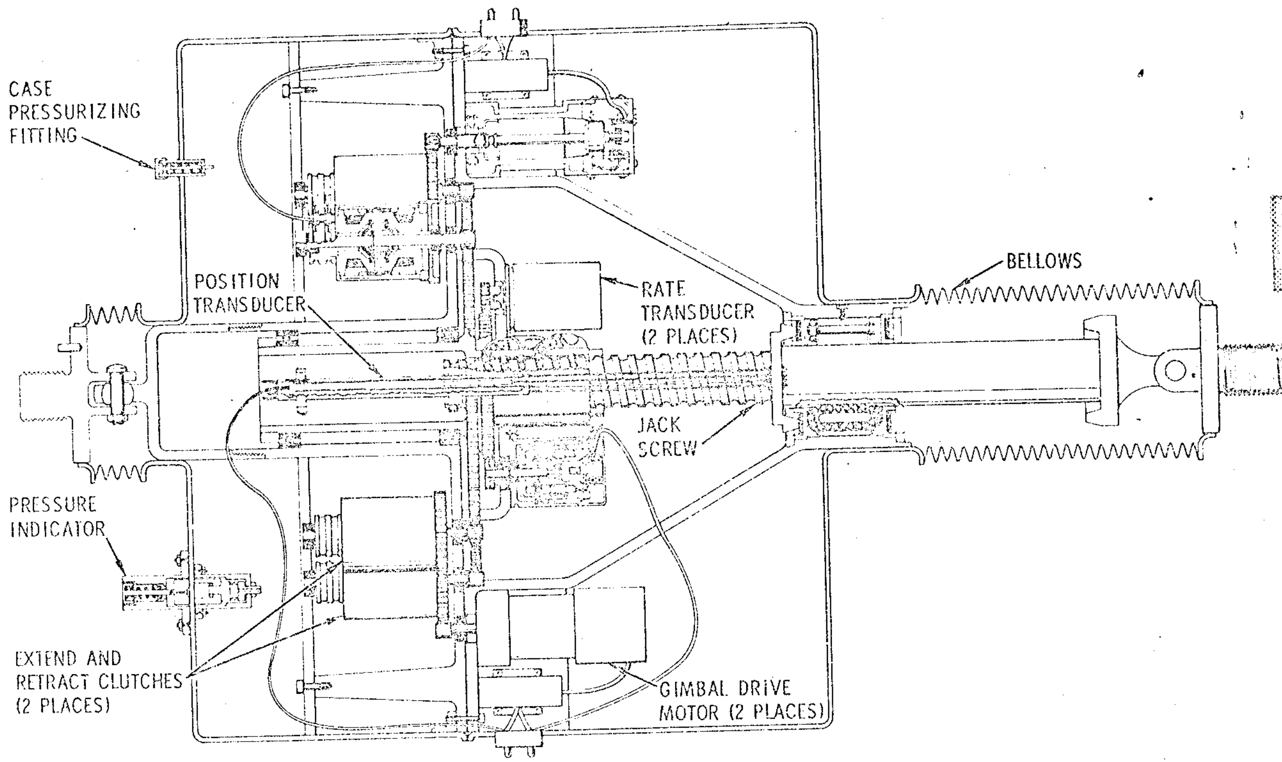


Figure 2.4-3. SPS Electro-Mechanical Gimbal Actuator

One motor and a pair of clutches (extend and retract) are identified as system No. 1, and the remaining motor and pair of clutches (extend and retract) are identified as system No. 2 within the specific actuator.

An over and undercurrent relay is employed for each primary gimbal motor (figure 2.4-4), and is controlled individually by a switch on panel 3. When the primary GIMBAL MOTORS switches are placed to the START position, power is applied to the motor-driven switch within the over and undercurrent relay of the primaries. The motor switch then supplies power from the main bus A to the gimbal motor. When the switch is released, it spring-loads to the ON position which activates the over and undercurrent sensing circuitry of the primary relay which monitors the current to the gimbal motor.

The over and under current relay of the primaries are utilized to monitor the current to the gimbal motor due to the variable current flow to the gimbal motor that is dependent upon the gimbal angle change required.

Using No. 1 yaw system as an example, and identifying the upper motor and clutches in figure 2.4-3 as system No. 1, the operation of the current monitoring system is as follows:

a. If the relay senses an over or undercurrent to gimbal motor No. 1, the monitor circuitry within the relay will drive the motor-driven switch, removing power from gimbal motor No. 1.

b. Simultaneously, a fail sense signal is sent from the relay, the SCS, which opens relay K13 removing inputs from the No. 1 clutches and closes relay K14 applying inputs to the No. 2 clutches within the same actuator; and the top contact of K12 opens and the bottom contact of K12 closes for TVC monitor.

c. Simultaneously a signal is sent to illuminate a caution and warning light on panel 11, to indicate the primary gimbal motor has failed.

The primary switches on panel 3 are then placed in the OFF position. Normally the OFF position is used to shut down the gimbal motors upon the completion of a thrusting period.

The No. 2 systems employ a 70-amp circuit breaker that is located in the service module near the SPS power distribution box.

Using No. 2 yaw system as an example (figure 2.4-4), the operation of the current monitoring system is as follows:

a. When the secondary GIMBAL MOTORS switches are placed to the START position, power is applied to the motor-driven switch within the over and undercurrent relay of the secondaries. The motor switch then supplies power from the main bus B through the 70-amp circuit breaker, and through the motor-driven switch to the secondary gimbal motor.

b. When the secondary switch is released, it spring-loads to the ON position which performs no functions on the secondaries.

c. The 70-amp circuit breaker will monitor the current to the secondary gimbal motor; and if a current of 70 amps is sensed, the circuit breaker in the service module will remove power from the secondary gimbal motor.

d. There is no fail sense signal sent to the stabilization and control system, and no illumination of the caution and warning light on panel 11 from the secondary system.

e. If the No. 2 system has failed due to an overcurrent, the circuit breaker removes power from gimbal motor No. 2 only, and that specific actuator is inoperative if the No. 1 system has previously failed.

The secondary switches on panel 3 are then placed to OFF position. Normally, the OFF position is used to shut down the gimbal motors upon the completion of a thrusting period.

The clutches are of a magnetic particle type. The gimbal motor drive gear meshes with the gear on the clutch housing. The gears on each clutch housing mesh and as a result, the clutch housings counterrotate. The current input is applied to the electro-magnet mounted to the rotating clutch housing from the stabilization control system or in the guidance and navigation system through the stabilization control system. A quiescent

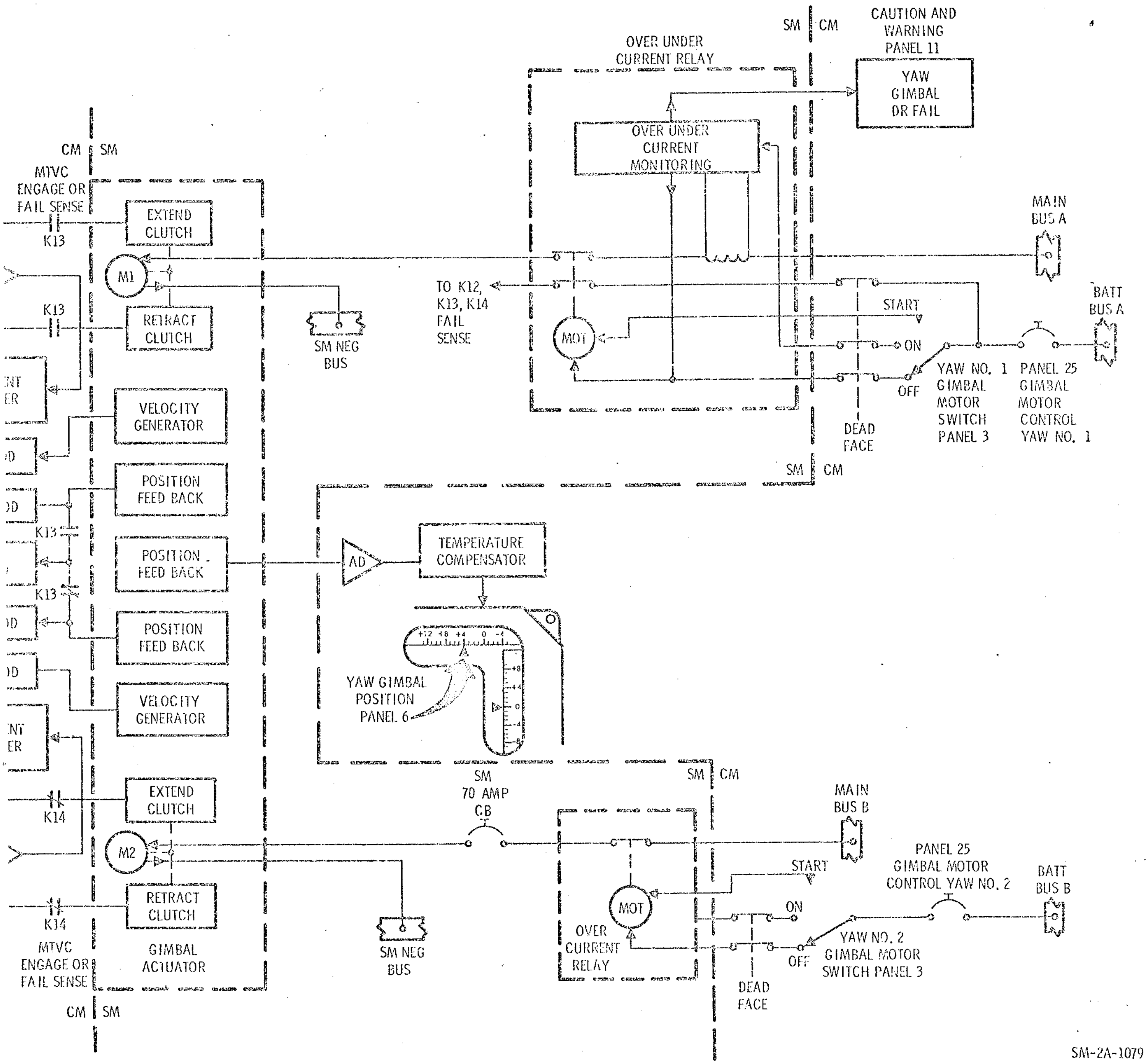
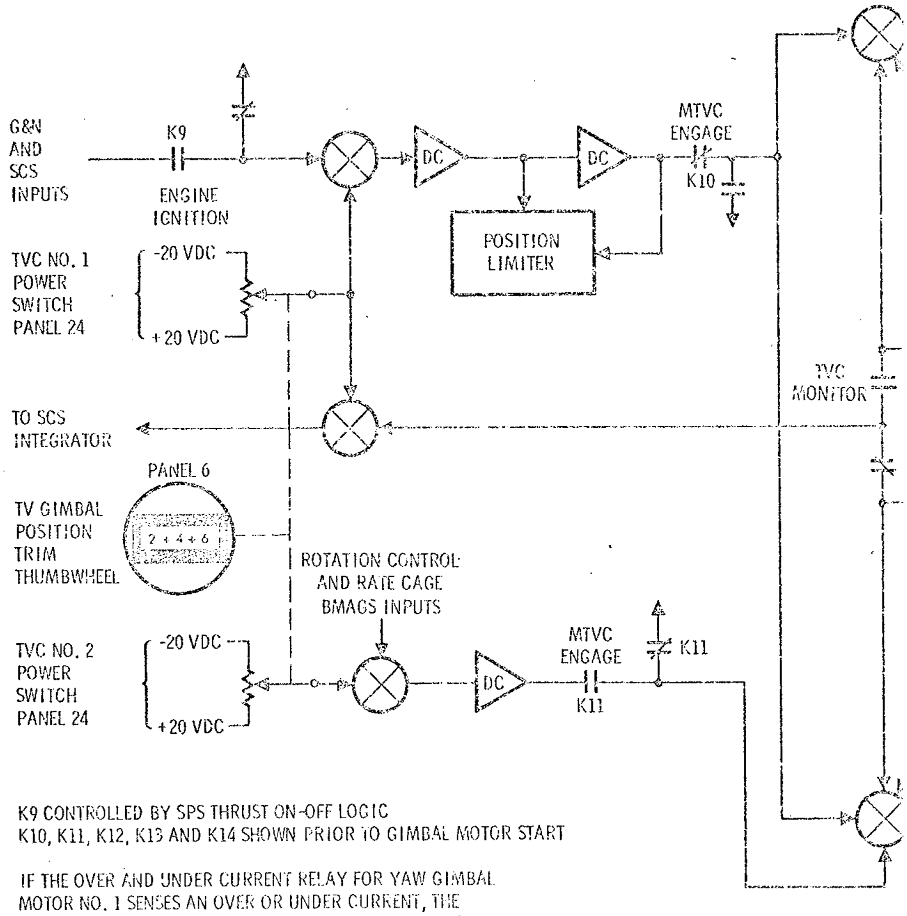


Figure 2.4-4. SPS Yaw Gimbal Actuator Motor and Clutch Control



K9 CONTROLLED BY SPS THRUST ON-OFF LOGIC
K10, K11, K12, K13 AND K14 SHOWN PRIOR TO GIMBAL MOTOR START

IF THE OVER AND UNDER CURRENT RELAY FOR YAW GIMBAL MOTOR NO. 1 SENSES AN OVER OR UNDER CURRENT, THE RELAY SIGNAL OPENS K13 TO CLUTCHES, K13 TOP CONTACT OPENS LOWER CLOSES FOR TLM, K13 CLOSES TO CLUTCHES, K12 TOP CONTACT OPENS, LOWER CLOSES.

K10, K11, K13 and K14 ARE ALSO CONTROLLED BY CLOCKWISE ROTATION OF TRANSLATION CONTROL WITH DELTA V MODE SELECTED REFERRED TO AS MANUAL THRUST VECTOR CONTROL current is applied to the electro-magnet of the extend and retract clutches when the thrust vector POWER switches, on panel 24, are on, which prevents the engine from moving during the boost phase of the mission with the gimbal motors OFF. A 246 foot-pound force would be required on the engine to overcome the quiescent current (equivalent to 1.53 g) in the clutches. The gimbal motors will be turned on, prior to jettisoning the launch escape tower, to support the SPS abort criteria after the launch escape tower has been jettisoned, and will be turned OFF as soon as possible to reduce the heat problem that occurs due to the gimbal motors driving the clutch housings. With quiescent current applied, it creates a friction force in the clutch housing that creates heat, which if allowed to rise to a high temperature the electro-magnet would loose its magnetism capability; thus rendering that set of clutches inoperative. This problem occurs only during boost, not during the normal SPS burns.

Prior to any thrusting periods the thumbwheels on panel 6 can be used to position the engine. The thrust-on signal may be provided by the G&N through the SCS, or by SCS only. In either mode the current input required (to maintain the engine thrust vector through the center of mass) to the clutches will increase above the quiescent current which increases the current in the electro-magnets that are rotating with the clutch housings. The dry powder magnetic particles have the ability to become magnetized very readily as well as becoming demagnetized as readily. The magnetic particles increase the friction force between the rotating housing and the flywheel, causing the flywheel to rotate. The flywheel arrangement is attached to the clutch output shaft allowing the clutch shaft to drive the bull gear. The bull gear drives a ball nut which drives the actuator jack-shaft to an extend or retract position, depending upon which clutch housing electro-magnet the current input is supplied to. The larger the excitation current, the higher the clutch shaft rotation rate.

Meshed with the ball nut pinion gear are two rate transducers of the tachometer type. When the ball nut is rotated, the rate transducer supplies a feedback into the summing network of the thrust vector control logic to control the driving rates of the jackscrew (acting as a dynamic brake to prevent over or undercorrecting), one transducer for each system.

The jackscrew contains four position transducers, all arranged for linear motion and all connected to a single yoke. One of the position transducers is used to provide a feedback to the summing network of transducer feedback. The two feedbacks to the summing network reduce the output current to the clutch, resulting in a proportional rate change to the desired gimbal angle position and returns to a quiescent current.

One position transducer provides a signal to the visual display on panel 6. One of the position transducers provides a feedback to the redundant summing network of the thrust vector logic for the redundant clutches. The remaining position transducer is not presently utilized.

The spacecraft desired motion, thumbwheel positioning, engine nozzle position, thrust vector position, gimbal position display indicator and actuator ram movement is identified in figure 2.4-5.

A snubbing device provides a hard stop for an additional 1-degree travel beyond the normal gimbal limits.

Manual thrust vector control may be obtained by the crew placing the translation control clockwise. This will open contacts K10 and K13 and close contacts K11 and K14, allowing the rotation control to provide the crew with manual thrust vector control.

2.4.2.10 Propellant Utilization and Gauging Subsystem (PUGS).

The subsystem consists of eight propellant quantity sensors, a propellant utilization valve, a control unit, and a display unit (figures 2.4-6 and 2.4-7).

2.4.2.10.1 Quantity Sensing, Computing, and Indicating System.

Propellant quantity is measured by two separate sensing systems: primary and auxiliary. The primary quantity sensors are cylindrical capacitance probes, mounted axially in each tank. In the oxidizer tanks, the probes consist of a pair of concentric electrodes with oxidizer used as the dielectric. In the fuel tanks, a pyrex glass probe, coated with silver on the inside, is used as one conductor of the capacitor. Fuel on the outside of the probe is the other conductor. The pyrex glass itself forms the dielectric. The auxiliary system utilizes point sensors mounted at intervals along the primary probes to provide a step function impedance change when the liquid level passes their location centerline.

Primary propellant measurement is accomplished by the probes capacitance being a linear function of propellant height.

Auxiliary propellant measurement is accomplished by locating the propellant level with point sensors. Each point sensor consists of concentric metal rings, seven in the storage tanks and eight in the sump tanks. The rings present a variable impedance, depending on whether they are covered or uncovered by the propellants. When the propellants are between point sensors, the propellants remaining are integrated by a rate flow generator which integrates the servos at a rate proportional to the normal flow rate of the fuel and oxidizer. A mode selector senses when the propellant crosses a sensor and changes the auxiliary servos from the flow rate generator mode to the position mode, the system moves to the location specified by the digital-to-analog converter for 3/4 second to correct for any difference. The system then returns to the flow rate generator mode until the next point sensor is reached. Figures 2.4-8 and 2.4-9 illustrate point sensor locations that are covered by propellants. The nonsequential pattern detector functions to detect false or faulty sensor signals. If a sensor has failed, the information from that sensor is blocked from the system, preventing disruption of system computation.

S/C MOTION DESIRED BY THE CREW	THUMB WHEEL POSITIONED ON PANEL ASSEMBLY 6 TO A	ENGINE NOZZLE MOVES TOWARDS THE	THE THRUST VECTOR OF THE ENGINE MOVES TOWARDS THE	GIMBAL POSITION INDICATOR ON PANEL ASSEMBLY 6 MOVES TO A	ACTUATOR RAM
S/C NOSE DOWN (PITCH)	+ NUMERAL (UP)	+ Z AXIS OF THE S/C	- Z AXIS OF THE S/C	+ NUMERAL	PITCH EXTENDS
S/C NOSE UP (PITCH)	- NUMERAL (DOWN)	- Z AXIS OF THE S/C	+ Z AXIS OF THE S/C	- NUMERAL	PITCH RETRACTS
S/C RIGHT (YAW)	- NUMERAL (RIGHT)	+ Y AXIS OF THE S/C	- Y AXIS OF THE S/C	- NUMERAL	YAW RETRACTS
S/C LEFT (YAW)	+ NUMERAL (LEFT)	- Y AXIS OF THE S/C	+ Y AXIS OF THE S/C	+ NUMERAL	YAW EXTENDS

THE ACTUATOR NULL POSITION IN YAW IS 4° OFFSET FROM THE S/C X AXIS DUE TO THE ENGINE MOUNTING PADS IN THE SERVICE MODULE BEING CANTED (BLOCK I VEHICLES)

THE THRUST VECTOR IS 4° TO THE +Y AXIS OF THE S/C AND THE ENGINE NOZZLE IS 4° TO THE -Y AXIS OF THE S/C.

AS A RESULT THE GIMBAL POSITION INDICATOR ON PANEL ASSEMBLY 6 WILL INDICATE A +4° WHEN THE ACTUATORS ARE IN THE NULL POSITION.

TO PARALLEL THE THRUST VECTOR AND THE ENGINE NOZZLE WITH THE X AXIS OF THE S/C, THE YAW THUMB WHEEL ON PANEL ASSEMBLY 6 MUST BE POSITIONED TO 0° AND AS A RESULT THE YAW POSITION INDICATOR ON PANEL ASSEMBLY 6 WILL INDICATE 0°.

IN THIS CASE, THE YAW ACTUATOR IS NO LONGER IN ITS NULL POSITION IN RESPECT TO THE GIMBAL RING.

THE THRUST VECTOR AND THE ENGINE NOZZLE ARE NOW PARALLEL TO THE X AXIS OF THE S/C

THE ACTUATOR NULL POSITION IN PITCH IS 0° (BLOCK I VEHICLES).

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Figure 2.4-5. Gimbaling of the Service Propulsion Engine

When a THRUST-ON signal is provided with the SENSOR select switch in the PRIMARY or NORMAL position, the crew display digital readouts, and unbalance display will not change for 4.5±1.0 seconds to allow for propellant settling. However, TLM will receive the same signal as upon completion of the last firing after approximately one second of SPS THRUST-ON.

When the THRUST-ON signal is provided with the SENSOR select switch in AUXILIARY position, the crew display digital readouts and TLM will receive a change in information immediately which is generated from a flow rate integrator that simulates the nominal flow rate and transmits this as quantity information to the crew displays and TLM. The crew digital readouts, unbalance display, and TLM will not be updated to the propellant from a point sensor for 4.5±1.0 seconds after THRUST-ON. When the THRUST-ON signal is provided plus 4.5±1.0 seconds, if a point sensor is uncovered the crew digital readouts, unbalance display, and TLM will be updated to the propellant remaining at that point sensor. The time delay of 4.5±1.0 seconds is to the point sensor system and not to the auxiliary fuel and oxidizer servos, and is to allow for propellant settling.

Any deviation from the nominal oxidizer to fuel ratio (2:1 by mass) is displayed in pounds by the UNBALANCE indicator. The upper half of the indicator is marked INC, and the lower half is marked DEC to identify the required change in oxidizer flow rate to correct any unbalance condition.

When the SENSOR select switch is in the NORMAL position, the outputs of both sensor systems are continually compared in the comparator network. If a discrepancy occurs between total primary and total auxiliary fuel of 300 pounds, or a discrepancy between total primary oxidizer and auxiliary oxidizer of 300 pounds, the caution and warning indicator on panel 11 is illuminated. The output of the oxidizer sump tank servo amplifier and the primary potentiometer of the unbalance indicator are compared in the comparator network, and if 300 pounds or 90 percent of the critical unbalance indicated versus time remaining is reached (figure 2.4-10), the caution and warning light on panel 11 is illuminated.

With the SENSOR select switch in the PRIMARY position, the output of the oxidizer sump tank servo amplifier and the output of the primary potentiometer in the unbalance meter are compared in the comparator network and if 300 pounds or 90 percent of the critical unbalance indicated versus time remaining is reached, the caution and warning light on panel 11 is illuminated.

With the SENSOR select switch in the AUXILIARY position, the output of the auxiliary oxidizer servo amplifier and the output of the auxiliary potentiometer are compared in the comparator network, and if 300 pounds or 90 percent of the critical unbalance indicated versus time remaining is reached (figure 2.4-10), the caution and warning light on panel 11 is illuminated.

G, AND INDICATING SYSTEM

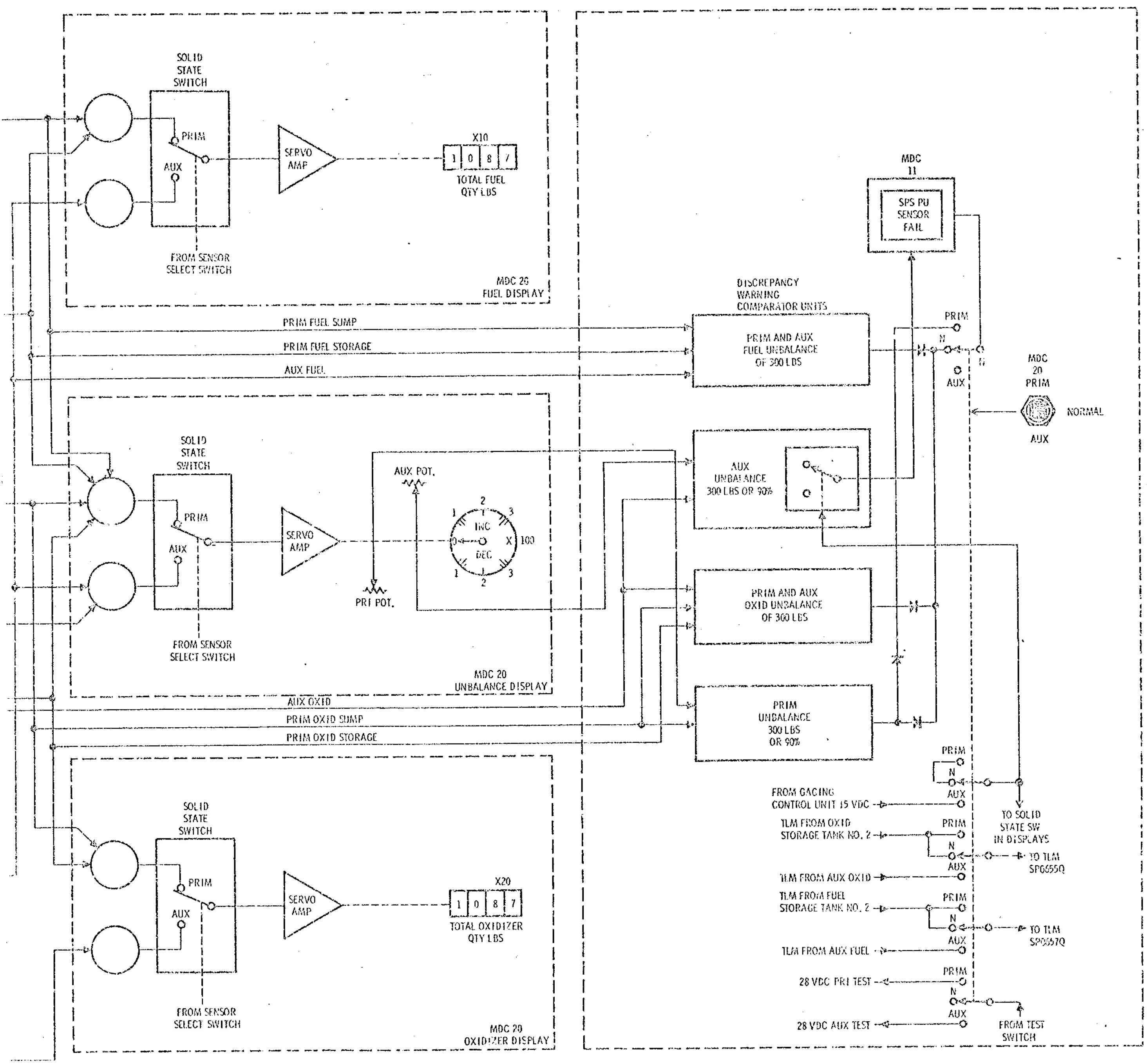

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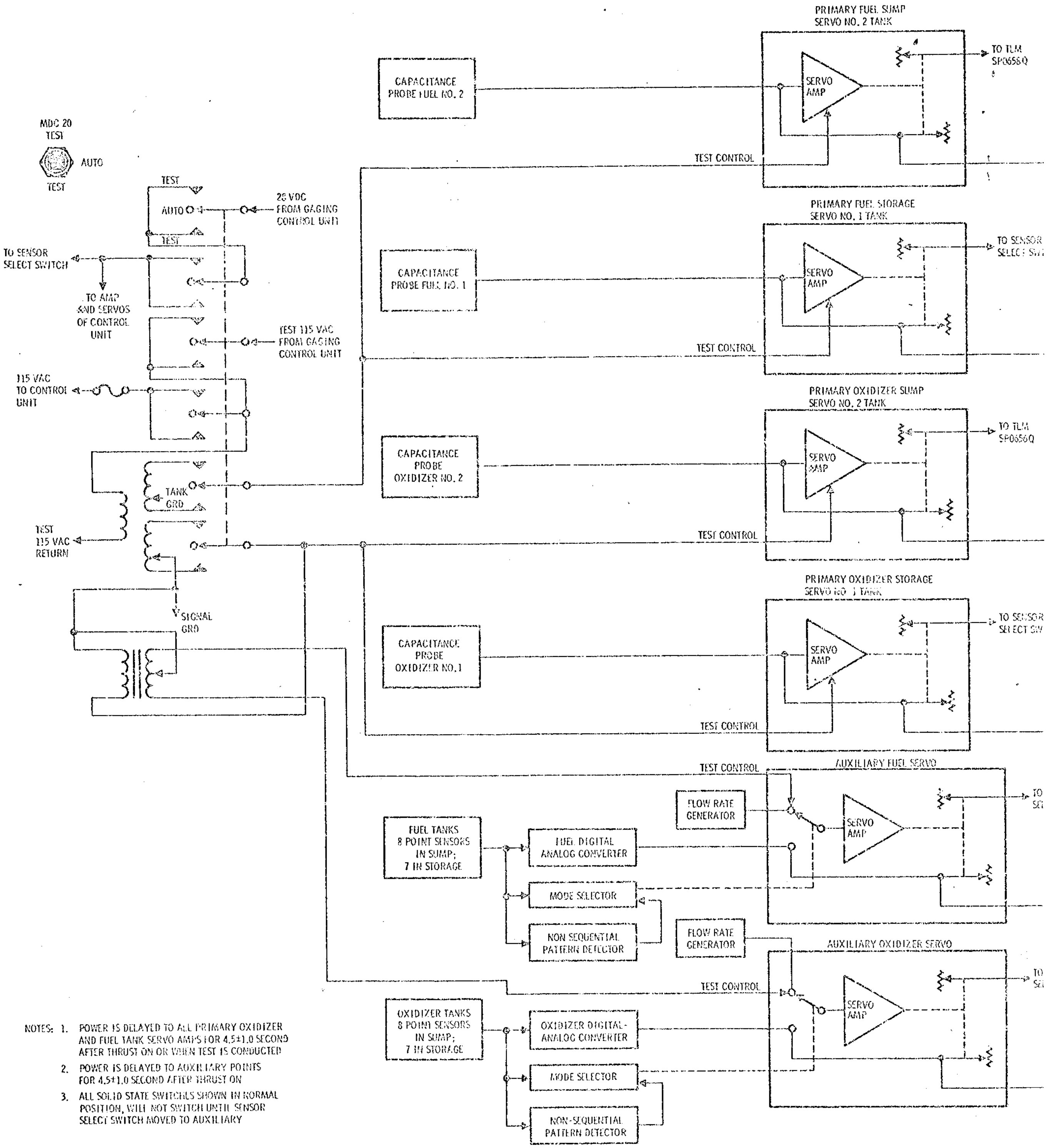
Figure 2.4-6. SPS Quantity Sensing Computing and Indicating System

SPS QUANTITY SENSING, COMP



NOTES:
1. POWER IS DELAYED TO ALL PRIMARY OXIDIZER AND FUEL TANK SERVO AMPS FOR 4.5±1.0 SECOND AFTER THRUST ON OR WHEN TEST IS CONDUCTED
2. POWER IS DELAYED TO AUXILIARY POINTS FOR 4.5±1.0 SECOND AFTER THRUST ON
3. ALL SOLID STATE SWITCHES SHOWN IN NORMAL POSITION, WILL NOT SWITCH UNTIL SENSOR SELECT SWITCH MOVED TO AUXILIARY

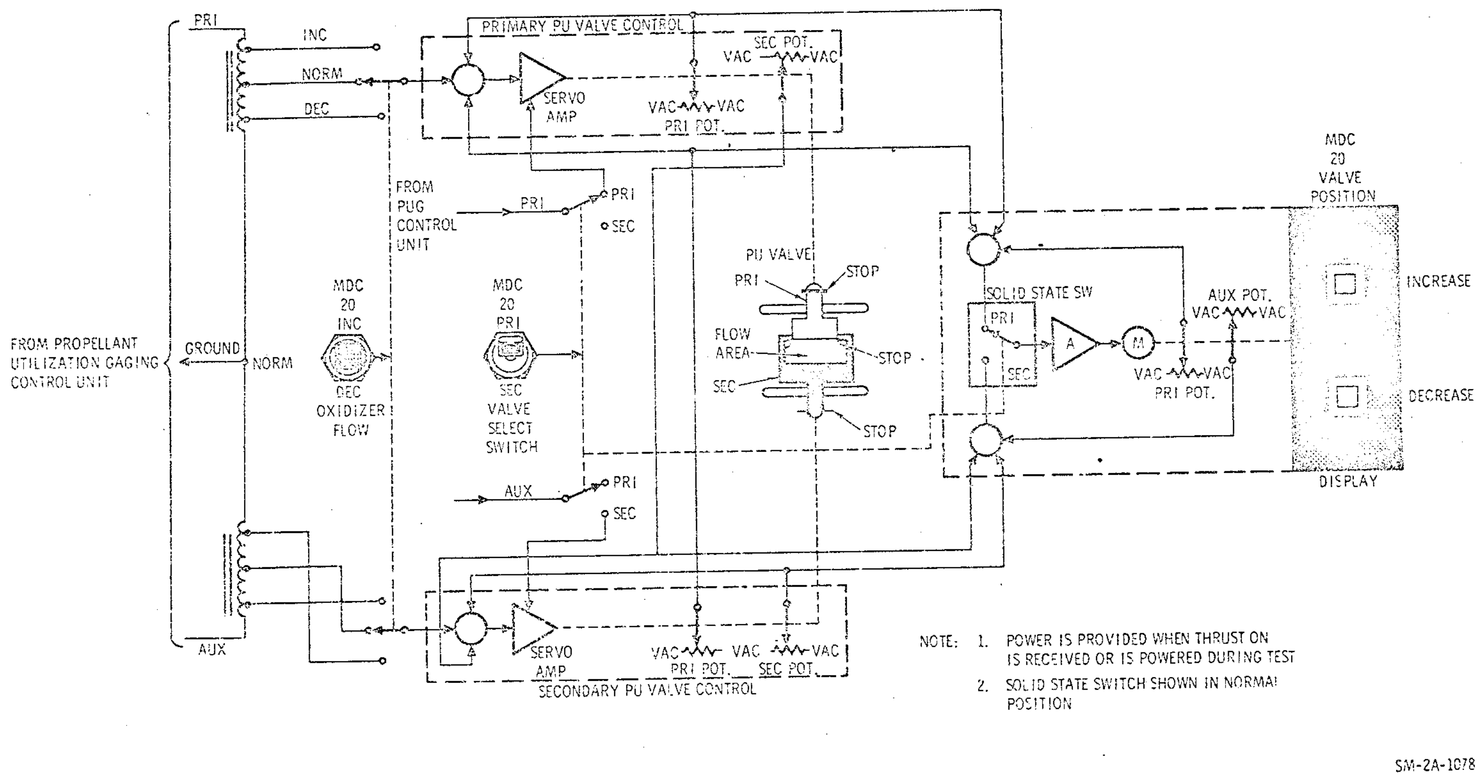


Figure 2.4-7. Propellant Utilization Valve Control and Flag Display.

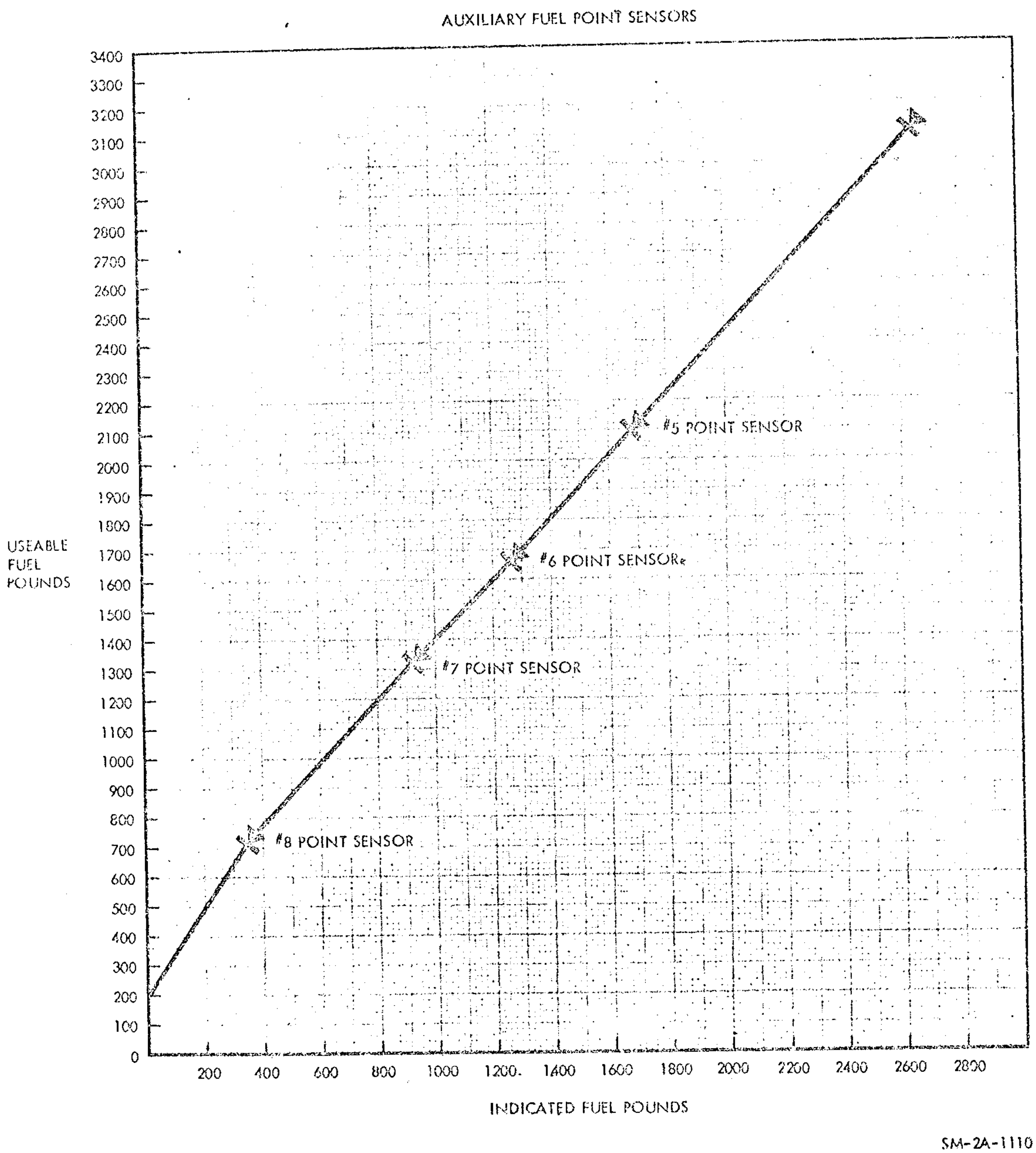


Figure 2.4-8. Auxiliary Fuel Point Sensors

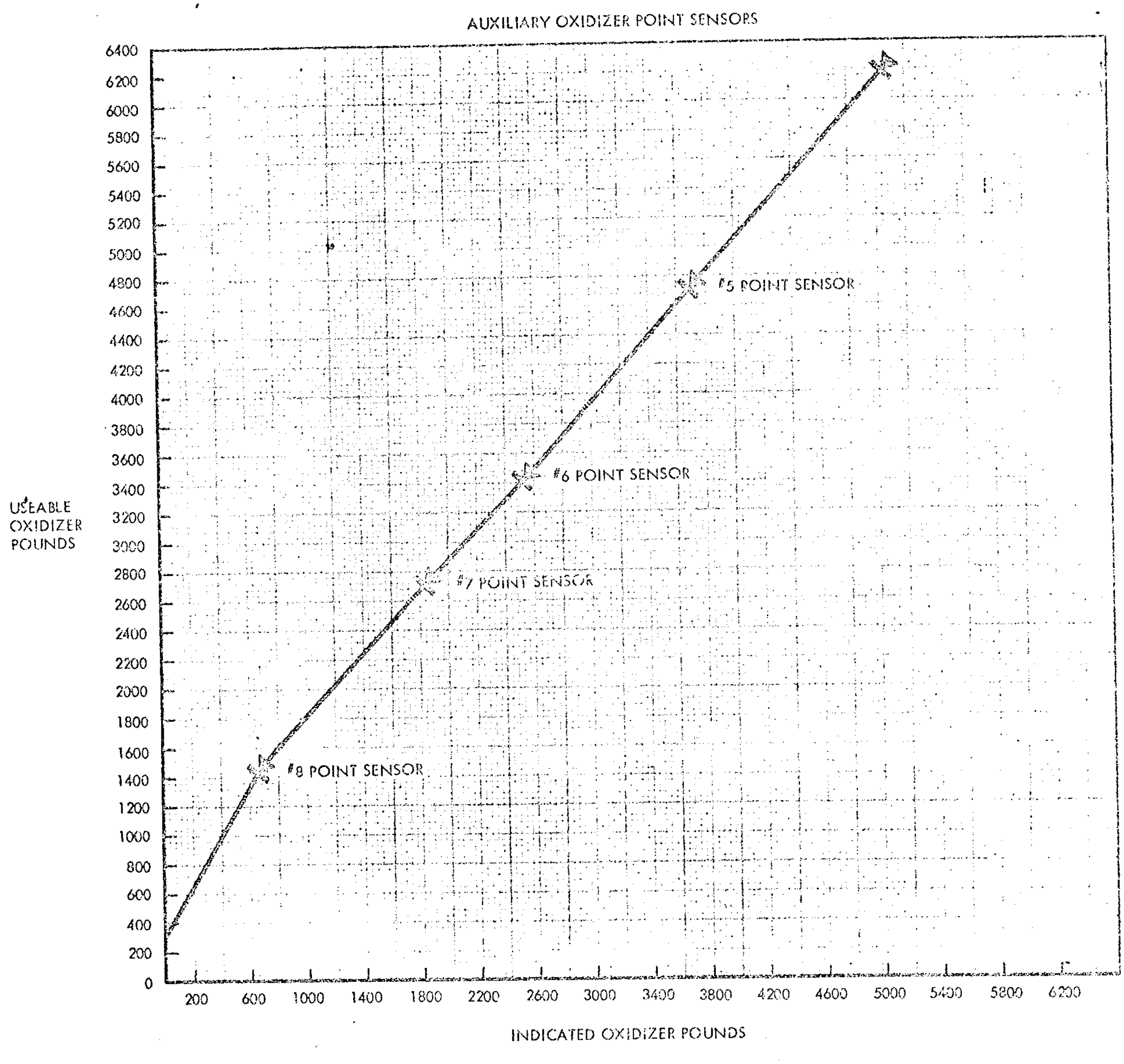


Figure 2.4-9. Auxiliary Oxidizer Point Sensors

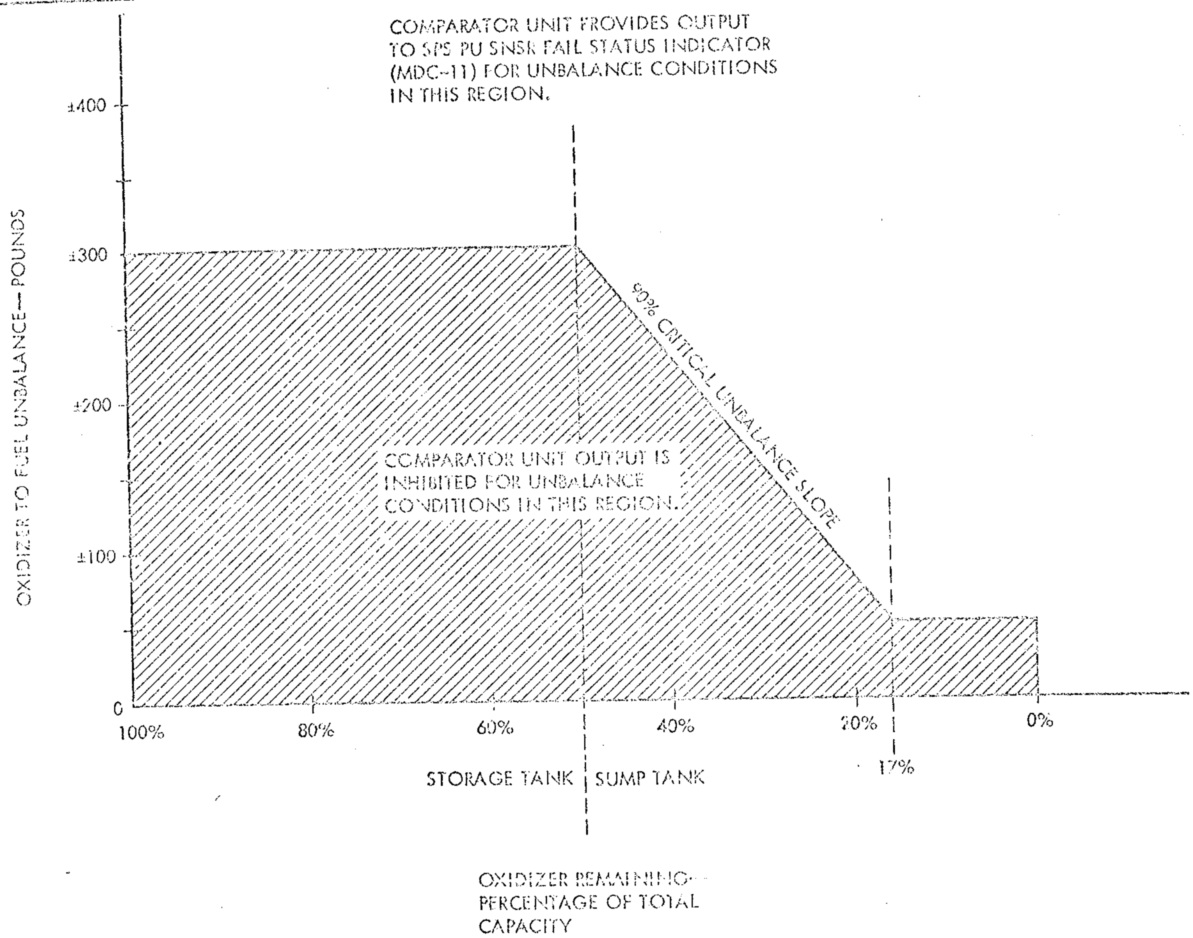


Figure 2.4-10. Propellant Unbalance Warning System Output Chart

Once the warning light is illuminated, the crew can determine whether there is a malfunction within the quantity and indicating systems or if there is a true unbalance condition existing by use of the self-test portion of the system. By observing the response of each system in conjunction with the TEST switch on panel 20, the crew can recognize the malfunction or determine if there is a true unbalance existing.

2.4.2.10.2 Quantity Computing and Indicating System Test.

A test of the sensing systems excluding the point sensors and probes can be implemented during THRUST-ON or -OFF periods.

With the SENSOR select switch in PRIMARY and the TEST switch in the TEST UP position, the test stimuli is applied to the primary system tank servo amplifiers after a 4.5±1.0 second delay. At this time, the test stimuli will drive the fuel and oxidizer displays to an increase reading at different rates (oxidizer at a rate of approximately 3 digits per second and fuel at a rate of approximately 1.5 digits per second), resulting in an unbalance and indicated on the unbalance display as an INC (clockwise rotation). TLM would receive an increase in propellant quantity from the primary system simultaneously.

Placing the TEST switch in the TEST DOWN position, and after a 4.5±1.0-second delay, will provide test stimuli to the tank servo amplifiers which will drive the fuel and oxidizer displays to a decrease reading, returning the crew displays close to the reading displayed prior to TEST UP, simultaneously TLM would receive a decrease in propellant quantity. If the TEST switch was again placed to TEST DOWN, after a 4.5±1.0-second delay, the fuel and oxidizer crew display readouts would drive to a decrease reading at different rates resulting in an unbalance and indicated on the unbalance display as a DEC (counterclockwise rotation). TLM would receive a decrease in propellant quantity simultaneously. To return to the reading displayed prior to the second TEST DOWN, place the TEST switch to TEST UP and after a 4.5±1.0-second delay, the crew displays would return close to the original displayed readings, simultaneously TLM receives an increase in propellant quantity.

To test the auxiliary system, place the SENSOR select switch to AUXILIARY and utilize the TEST switch up and down positions. There is no time delay involved with the auxiliary system.

The AUTO position removes the electrical test stimuli inputs.

2.4.2.10.3 Propellant Utilization Valve.

If an unbalance condition exists, the crew will use the propellant utilization valve to return the propellants to a balanced condition. The propellant utilization valve housing contains two sliding gate valves within the housing. One of the sliding gate valves is the primary and the remaining is the secondary (figure 2.4-7).

Stops are provided within the valve housing for the full increase or decrease positions of the primary and secondary sliding gate valves.

The secondary propellant utilization valve has twice the travel of the primary propellant utilization valve to compensate for primary propellant utilization valve failure in any position. The secondary valve is selected by the VALVE switch and is controlled by the OXIDIZER FLOW switch in the same manner as the primary.

The propellant utilization valve controls are on panel 20. The VALVE switch selects the primary or secondary propellant utilization valve. Normal position of the VALVE switch is PRIMARY. The OXIDIZER FLOW switch is utilized to position the primary or secondary propellant utilization sliding gate valve. When the OXIDIZER FLOW switch is in NORMAL, the sliding gate valve is in the nominal flow position and the upper and lower OXID flow position indicator is gray. When the OXIDIZER FLOW switch is placed to the INCREASE position, the sliding gate valve is in the increase flow position and the upper OXID flow position indicator is striped. When the OXIDIZER FLOW switch is placed to DECREASE position, the sliding gate valve is in the decrease flow position, but does not block the oxidizer flow completely and the lower OXID flow indicator is striped.

2.4.2.10.4 Engine THRUST ON-OFF Control.

Figure 2.4-1 illustrates the THRUST ON-OFF logic in the guidance and navigation delta V mode, the stabilization control system delta V mode and the manual direct mode.

The guidance and navigation is the primary delta V mode which provides the most accurate trajectory corrections. The G&N ΔV mode of operation will require G&N gate 2 to be completely enabled. Its required inputs will be an ullage maneuver input which could be supplied by the SPS abort logic or the direct ullage pushbutton or by translation control 1 or 2 placed to the +X, which would satisfy the OR gate 3 function and the holding input after the G&N AND gate 2 is enabled which is processed through NAND gates 1 and 2, with the G&N ΔV mode selected from the SCS control panel and the pulse train converter output of logic one commanded from the G&N computer by the crew. These inputs will enable AND gate 2 and provide the logic one input to inverter 3 which disables AND gate 6. Inverter 4 will provide a logic one signal to the solenoid drivers that provide the ground for the two sets of SPS relays. The two sets of SPS relays provide power to the following:

a. The four solenoid control valves, which allow gaseous nitrogen pressure to be directed to four actuators that position eight of the ball valves in the injector valve assembly of the engine. This is due to INJECTOR PRE VALVE A and B being enabled.

b. The quantity sensing and indicating system and the propellant utilization valve.

c. The systems A and B helium pressurizing valves.

d. When the output of the pulse converter is a logic zero, G&N AND gate 2 is disabled, which terminates a G&N ΔV maneuver and removes ground from the two sets of SPS relays. (Manual backup of the THRUST CONTROL switch to OFF.)

The backup delta V mode, is the stabilization control system delta V mode which has limitations and restrictions that require individual consideration. The SCS ΔV mode of operation will require enabling SCS AND gate 1 to initiate thrusting of the engine. SCS AND gate 1 has three enabling inputs: the input from OR gate 1 indicating a ΔV mode is selected; the output from the DECA indicating a value greater than zero on the ΔV REMAINING display; and the input from the THRUST ON pushbutton light indicator after it is processed through NAND gates 3 and 4. The one logic output from SCS AND gate 1 is inverted by INVERTER 2 to a zero logic which disables AND gate 6 and is inverted by INVERTER 4 to a logic one, and enables the solenoid drivers that provide ground to the two sets of SPS relays. The two sets of SPS relays provide power to the same functions as in the G&N ΔV mode.

a. When the output of the DECA ΔV REMAINING counter reaches zero velocity, SCS AND gate 1 is disabled. The logic output from AND gate 1 goes to zero and INVERTER 2 goes to a one logic output which enables AND gate 6 to a one logic output and the signal is inverted by INVERTER 4 to a zero logic output, and the solenoid drivers remove ground from the two sets of SPS relays terminating thrust. (Manual backup of the THRUST CONTROL switch to OFF.)

The DIRECT operating mode bypasses all electronics. The DIRECT ON position of the THRUST CONTROL switch provides power to the two sets of SPS relays, the same as in the G&N ΔV mode of SCS ΔV mode.

Thrust would be terminated by placing the THRUST CONTROL switch in the OFF position.

The SPS thrust control logic provides illumination of the talk-back light in the THRUST ON switch. To illuminate the light in a G&N ΔV mode, AND gate 4 must be enabled. It has two inputs which are G&N ΔV mode selected and the presence of a logic one output from the pulse train converter. With AND gate 4 enabled, OR gate 2 will trigger the lamp driver and illuminate the THRUST ON light. To illuminate the light in the SCS ΔV mode, AND gate 3 must be enabled, it will also satisfy OR gate 2 and trigger the lamp driver. The SPS thrust control logic is interlocked so that AND gates 3 and 4 will never be enabled simultaneously because of the THRUST ON switch inhibited to the pulse train converter. Therefore, the THRUST ON light will be illuminated at all times when the thrust is on for normal operation. If the direct function is used to initiate thrusting, it will bypass all electronics and not illuminate the THRUST ON light.

The output from the SPS thrust control logic performs a function other than energize the fuel and oxidizer solenoid valves. It provides the logic switches required to reconfigure the SCS relays for proper thrust vector control. The thrust control switch provides both inputs to OR gate 4 if both dc buses are operational. Either dc bus will enable OR gate 5 to provide the upper input to INHIBIT AND gate 1. AND gate 5 provides the controlling input to the INHIBIT AND gate 1. When the solenoid drivers are not energized and the thrust control switch is in NORMAL, both inputs to AND gate 5 are true. The true input will maintain a logic false output from the INHIBIT AND gate 1 because of the inversion on the input of the gate. When either SPS relay set is activated by a solenoid driver or by the ground contacts of the DIRECT ON switch, AND gate 5 will be disabled by the ground or false input. The input will be inverted by the INHIBIT AND gate 1 input to enable an output to the time delay. The time delay is required to permit thrust buildup from the SPS engine before the ullage maneuver is terminated; however, the pitch and yaw attitude error inputs are inserted into the TVC electronics immediately. This action assures positive g forces throughout thrust initiation of the SPS engine and gimbaling control of the SPS engine for the entire thrust buildup. The time delay is also active after INHIBIT AND gate 1 is disabled by enabling AND gate 5. AND gate 5 is enabled when both sets of SPS relays are de-energized. The time delay is active for both outputs when the signal is removed to permit continued SPS engine gimbal control, while the residual thrust is present and inhibits RCS operation in pitch and yaw until residual thrust is reduced. This reduces the tumbling induced at thrust termination. The time delay is approximately one second for application and removal of the engine ignition signal.

2.4.3 PERFORMANCE AND DESIGN DATA.

2.4.3.1 Design Data.

The following list contains specific data for the components in the SPS:

HELIUM TANKS (2)	4000±50 psia fill pressure, 4400 maximum operating pressure 70°±10°F, capacity 19.4 cubic feet, inside diameter 40 in. and a wall thickness of 0.46 in.
REGULATOR UNITS (2)	Working regulator - Primary 186±4 psig, secondary 191±4 psig, primary lockup 200 psig, secondary lockup 205 psig. Normally locked by working regulator; primary 181±4 psig, secondary 191±4 psig, primary lockup 195 psig, secondary lockup 205 psig.
PRESSURE TRANS-DUCERS (2) ULLAGE PRESSURE (REGULATED HELIUM)	Fuel and oxidizer underpressure setting (SPS PRESS light, panel 11) 160 psia. Fuel and oxidizer overpressure setting (SPS PRESS light, panel 11) 200 psia.
PROPELLANT UTILIZATION VALVE CONTROL (2)	Increase position - 46.65 lbs/second Normal position - 45.27 lbs/second at 70°F and 168±4 psig Decrease position - 43.87 lbs/second Response time - Normal to increase or vice-versa, or normal to decrease or vice-versa is 3 to 4 seconds
QUANTITY SENSING SYSTEM ACCURACY	±0.35% of full tank plus ±0.35% of propellant remaining primary system ±0.35% of full tank plus ±0.35% of propellant remaining plus 2.3% of storage tank quantity remaining, auxiliary system.

HELIUM RELIEF VALVE (2)	Diaphragm ruptures at 220±7 psig Filter - 10 microns normal, 25 microns absolute Relief valve relieves at 232±8 psig Relief valve reseats at 212 psig minimum Flow capacity 3 lbs/second minimum at 60 F and 250 psig Bleed device closes at ambient to 100 psi and is manually reset open
OXIDIZER STORAGE AND SUMP TANK	Total tank capacity 30, 600 lbs, usable 27, 333 lbs. Each tank has a volume of 175 cubic ft. Fill pressure 110 psia. Ullage after filling 2.4 cubic ft in storage and 8.0 cubic ft in sump tank. Ullage after pressurized to 175 psia, 6.8 cubic ft in storage and 5.0 cubic ft in sump tank. Inside diameter 51 in., length 165.4 in., and will elongate to 0.125 in. when pressurized to 240 psi and 120°F for 336 hours. Wall thickness 0.060 in. in continuous areas, 0.069 in. weld buildup areas and 0.031 in. on domes. Fill tolerance of 1/4 of 1% of full tank plus 1/4% of propellant remaining.
FUEL STORAGE AND SUMP TANK	Total tank capacity 15, 300 lbs, usable 13, 677 lbs. Each tank has a volume of 139.7 cubic ft. Fill pressure 90 psia. Ullage after filling 1.8 cubic ft in storage and 5.8 cubic ft in sump tank. Ullage after pressurized to 175 psia, 6 cubic ft in storage and 3.0 cubic ft in sump tank. Inside diameter 45 in., length 166.8 in., and will elongate to 0.125 in. when pressurized to 240 psi and 120°F for 336 hours. Wall thickness 0.053 in. in continuous areas, 0.061 in. in weld buildup areas, and 0.031 in. on domes. Fill tolerance of 1/4 of 1% full tank plus 1/4% of propellant remaining.

NOMINAL PROPELLANT TANK WORKING PRESSURE	175±4 psia
INTERFACE FLANGE FILTER	500 microns absolute
GASEOUS NITROGEN PROPELLANT VALVE CONTROL SYSTEMS (2)	Storage vessel pressure 2500±50 psi at 68°F, 2900 psi at 130°F. Support 36 valve actuations.
	Regulator, single stage, 130±7 psi at all flow up to 0.035 lbs/second.
	145 psi maximum lockup pressure.
	Relief valve - relieves at 350±10 psi, reseats at not less than 250 psi
	Ball valves 1 and 4 dry opening travel time of 0.6 (+0.2, - 0.05) second
	Ball valves 2 and 3 dry opening travel time of 0.325±0.1 second.
	Ball valves 1 and 4 dry closing travel time of 0.375±0.05 second
	Ball valves 2 and 3 dry closing travel time of 0.575±0.1 second
ENGINE	500-second service lift on S/C 014
	Capable of 36 restarts
	Expansion ratio, 6 to 1 at ablative chamber exit area, 62.5 to 1 at nozzle extension exit area.
	Cooling chamber, ablation and film extension, radiation
	Injector type, baffled regeneratively, cooled, unlike impingement
	Oxidizer lead, 8 deg.
	Length, 152.82 inches

	Nozzle extension exit diameter, 94.4 inches
	Weight, approximately 650 lbs
	Ablative chamber throat temperature sensor, illuminates SPS WALL TEMP HI caution and warning light on panel 11 at 380°F, one sensor per harness.
	SPS P_c transducer, P_c displayed on panel 3 through L/V AOA, SPS P_c switch to L/V AOA, SPS P_c indicator, green range on indicator is 65 to 125%, normal P_c 85 to 125 psia.
GIMBAL LIMITS	About Z-Z axis ±7 (+1/2, -0) deg with additional 1 deg for snubbing yaw
	About Y-Y axis ±6 (+1/2, - 0) deg with additional 1 deg for snubbing pitch
GIMBAL MOTOR UNDER AND OVERCURRENT RELAYS	Undercurrent (primary only) below 6 amperes detected for a duration of 250±50 milliseconds or more shall interrupt the flow to the load in less than 100 milliseconds.
	Overcurrent dependent upon temperature during start transient and steady state of gimbal motor on primary. Secondary controlled by 70-amp circuit breaker.
ACTUATOR CLUTCHES	Quiescent current of 60 (+10, -5) milliamps (Motors off, a 246 ft-lb force required to move engine equivalent to 1.53 g.)
ACTUATOR PRESSURIZED S/C 012 and 014 SERVO ACTUATOR DEFLECTION RATE	3.5±1.0 psi dry air at -160°F at vendor 0.23 radians per second (13.09° per second)
FLIGHT COMBUSTION	180 g's peak to peak for 70±20 milliseconds
STABILITY MONITOR SYSTEM	600 to 5000 cycles per second

2.4.3.2 Performance Data.

Refer to mission modular data book, SID 66-1177.

2.4.3.3 Power Consumption Data.

Subsystem and Component	Control	No. of Units	Watts per Unit AC	Watts per Unit DC	Total Watts AC	Total Watts DC
Service propulsion Propellant utilization valve	VALVE sw and OXIDIZER FLOW sw (Ref. gauging sys)				62.6	16.8
Propellant utilization gauging system	Gauging CB(4) SPS GAUGING AC sw SENSOR sw					
Injector prevalves	He valve CB(2) INJECTOR PREVALVE sw (2)	2		21.0		42.0
He solenoid valves	He PRESSURIZING sw (2) SPS relay	2		28.0		56.0
Feedline heaters	SPS HEATER sw	26		(See figure 2.4-2)		33.0
Pilot valves (SCS)	ECA drivers (2)	4		10.5		42.0
Gimbal actuator motors	Gimbal CB (6) GIMBAL MOTOR sw (4)	Channel I				
		Idle				
		pitch		450*		900
		yaw		450*		
		Boost				
		pitch		511*		1022
		yaw		511*		
		Thrust ON				
		pitch		775*		1550
		yaw		775*		
		Maximum				
		pitch		1800*		3600
		yaw		1800*		
		Channel II				
		Idle				
		pitch		335**		670
		yaw		335**		
		Boost				
		pitch		335**		670
		yaw		335**		
		Thrust ON				
		pitch		335**		670
		yaw		335**		
		Maximum				
		pitch		335**		670
		yaw		335**		

*With quiescent current
**Without quiescent current

Above statements assume channel I is the operating channel and channel II is standby. Channel I operating values applicable to channel II when channel II is commanded. 28-vdc supply values, current draw values not a direct function of an applied voltage.

2.4.3.4 SPS Electrical Power Distribution.

See figure 2.4-11 for electrical power distribution.

2.4.4 OPERATION LIMITATIONS AND RESTRICTIONS.

The operational limitations and restrictions of the SPS are as follows:

a. Propellant quantity gauging subsystem is operational only during engine firing. A 4.5-second firing period is required before propellant quantity information is updated, when SENSOR switch is in the NORM or PRI position. Delay is built-in to allow propellants to settle and stabilize before updating the displays.

b. A one-second time delay between actuation of GIMBAL MOTOR switches (MDC-3) is required, as simultaneous actuation may result in an excessive power drain.

c. Engine design minimum impulse control limit is 0.4 second; however, mission minimum impulse may be longer.

d. Due to adversive temperature effects, engine gimbal drive motors should not be continuously energized for periods in excess of 12 minutes.

e. Single bank mode of operation by the bipropellant valve assembly will result in a 3 percent reduction in thrust.

2.4.5 TELEMETRY MEASUREMENTS.

The following subsequent list is of all SPS telemetry data monitored by flight controllers and ground support personnel.

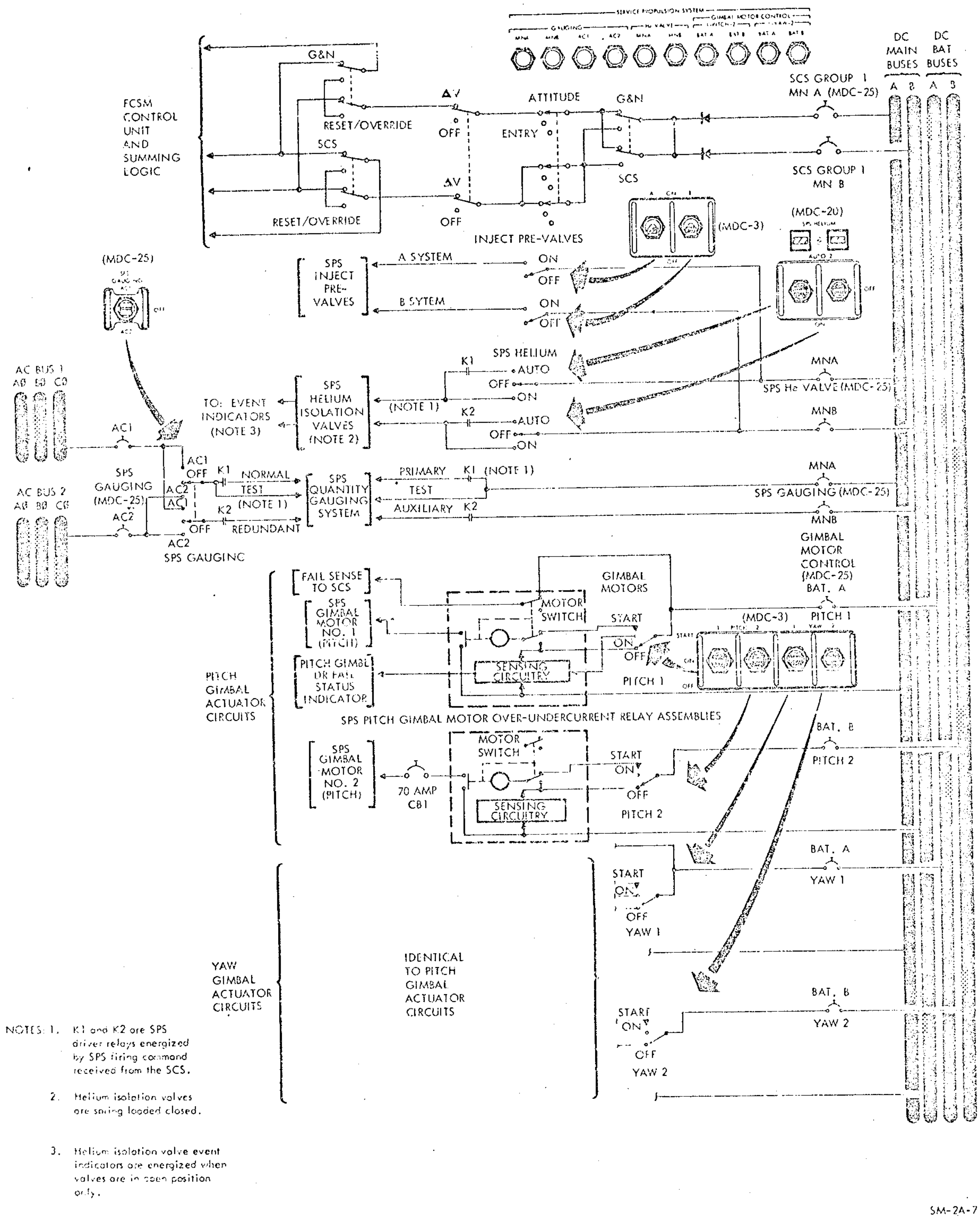


Figure 2.4-11. SPS Electrical Power Distribution Diagram

Crew Display	Figure	Measurement	Measurement Number	Range	Signal Conditioner	Channel Code*	Bit Rate*	*	Operating Range
Panel 20	2.4-1	Helium Tank Supply Pressure	SP0001P	+0 +5K psia	S28AR3	11A66	H2	PCM	4000±50 psia and decrease with firings
Panels 20 & 11	2.4-1	Regulated Helium Pressure to Fuel Tank	SP0006P	+0 +300 psia	S28AR6	11A70	H2	PCM	170 to 195 psia
Panels 20 & 11	2.4-1	Regulated Helium Pressure to Oxidizer Tank	SP0003P	+0 +300 psia	S28AR5	11A68	H2	PCM	170 to 195 psia
Panels 20 & 11	2.4-1 and 2.4-6	Quantity Sensing System							
		Sensor Select Switch in Normal							
		Primary Fuel Sump 2	SP0658Q	+0 +8K lbs		10A94	H1	PCM	
		Primary Fuel Storage 1	SP0657Q	+0 +8K lbs		10A92	H1	PCM	
		Primary Oxidizer Sump 2	SP0656Q	+0 +16K lbs		10A90	H1	PCM	
		Primary Oxidizer Storage 1	SP0655Q	+0 +16K lbs		10A88	H1	PCM	
		Sensor Select Switch in Primary							
		Primary Fuel Sump 2	SP0658Q	+0 +8K lbs		10A94	H1	PCM	Variable
		Primary Fuel Storage 1	SP0657Q	+0 +8K lbs		10A92	H1	PCM	
		Primary Oxidizer Sump 2	SP0656Q	+0 +16K lbs		10A90	H1	PCM	
		Primary Oxidizer Storage 1	SP0655Q	+0 +16K lbs		10A88	H1	PCM	
		Sensor Select Switch in Auxiliary							
		Primary Fuel Sump 2	SP0658Q	+0 +8K lbs		10A94	H1	PCM	
		Primary Oxidizer Sump 2	SP0656Q	+0 +16K lbs		10A90	H1	PCM	
		Auxiliary Fuel	SP0657Q	+0 +8K lbs		10A92	H1	PCM	
		Auxiliary Oxidizer	SP0655Q	+0 +16K lbs		10A88	H1	PCM	
	2.4-1	Temperature 1 Oxidizer Distribution Line	SP0054T	+0 +200°F					
	2.4-1	Temperature 2 Oxidizer Distribution Line	SP0055T	+0 +200°F					
	2.4-1	Temperature 1 Fuel Distribution Line	SP0057T	+0 +200°F					Variable
	2.4-1	Temperature 2 Fuel Distribution Line	SP0058T	+0 +200°F					
Panel 20	2.4-1	Fuel Inlet Pressure to Fuel Valve	SP0010P	+0 +300 psia	S28AR8	11A77	H2	PCM	135 to 195 psia
Panel 20	2.4-1	Oxidizer Inlet Pressure to Oxidizer Valve	SP0009P	+0 +300 psia	S28AR7	11A76	H2	PCM	135 to 195 psia
Panel 3	2.4-1	Chamber Pressure	SP0061P	+0 +150 psia	S28AR9	12A11	H1	PCM	65 to 125 psia
	2.4-1	Injector Temperature	SP0060T	+0 +200°F		11A109	H1	FQ	Variable
Panel 20	2.4-1	Primary Gaseous Nitrogen Tank Supply Pressure	SP0600P	+0 +5000 psia	S28AR41	10A78	H1	PCM	2500±50 psia and decrease with firings
Panel 20	2.4-1	Secondary Gaseous Nitrogen Tank Supply Pressure	SP0601P	+0 +5000 psia	S28AR42	11A125	H1	PCM	2500±50 psia and decrease with firings
Panel 20	2.4-1	Ball Valve Position 1	SP0022H	+0 +90 deg		11A103	H1	PCM	Open or closed
Panel 20	2.4-1	Ball Valve Position 2	SP0023H	+0 +90 deg		11A105	H1	PCM	Open or closed
Panel 20	2.4-1	Ball Valve Position 3	SP0024H	+0 +90 deg		11A110	H1	PCM	Open or closed